

1N - 10
63486

**Advanced Transportation System Studies
Technical Area 2 (TA-2)
Heavy Lift Launch Vehicle Development
Contract**

**NAS8-39208
DR 4**

Final Report

**Prepared by
Lockheed Martin Missiles & Space
for the
Launch Systems Concepts Office
of the
George C. Marshall Space Flight Center**

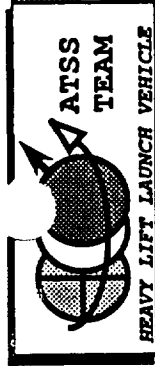
July 1995

Table of Contents

Section	Page
1.0 Introduction	1-1
2.0 SSTD Design Groundrules	2-1
3.0 Operations Issues and Lessons Learned	3-1
3.1 Operations Issues — Lessons Learned	3-1
3.2 Operations Issues — Requirements Flowdown	3-2
3.3 SSTD Operability Pros/Cons of Tripropellant Versus Bipropellant.....	3-3
4.0 Vertical-Takeoff/Landing Versus Vertical-Takeoff/Horizontal-Landing	4-1
4.1 VTOL/VTHL Pros/Cons Results Summary	4-1
4.2 Vertical Take-off/Vertical Landing Pros and Cons	4-2
4.3 Vertical Take-off/Horizontal Landing Pros and Cons	4-8
5.0 SSTD Design Results	5-1
5.1 Major Design Considerations.....	5-5
5.3 Technology Assumptions and Sizing Groundrules.....	5-7
5.4 Sizing Tool Description	5-17
5.5 Side Entry Conical VTOL Concept	5-19
5.6 Winged Body VTHL Concept	5-40
5.7 Lifting Body VTHL Concept.....	5-44
5.8 Study Conclusions.....	5-66
6.0 SSTD Simulation Results	6-1
6.1 Simulation Groundrules and Assumptions	6-1
6.2 Simulation Results	6-3
7.0 SSTD Assessment Conclusions	7-1
8.0 SSTD Sizing Tool User's Guide	8-1
9.0 SSTD Turnaround Assessment Report.....	9-1
10.0 Ground Operations Assessment First Year Executive Summary	10-1
11.0 Health Management System Definition Study	11-1
12.0 Major TA-2 Presentations	12-1
13.0 First Lunar Outpost Heavy Lift Launch Vehicle Design and Assessment Report.....	13-1
14.0 Russian Propulsion Technology Assessment Reports.....	14-1

10.0 Ground Operations Assessment First Year Executive Summary

This section contains a copy of an executive summary that was prepared by LSOC to document their support to the TA-2 contract during the first-year period of performance of the contract, May 1992 through May 1993. LSOC participated on the TA-2 contract as part of the concurrent engineering launch system definition team, and provided outstanding HLLV ground operations requirements and concept assessments for LMSC through an intercompany work transfer (funded by LMSC's TA-2 funding), as well as providing specific HLLV ground operations assessments at the direction of NASA KSC through KSC funding that was routed to the TA-2 contract. The LSOC activities were directed by Mr. Gary F. Letchworth of the LSOC Advanced Programs department, and principal support was provided by Mssrs. Steve Black, Steve Burns, Bob Seavey, and Ms. Arlene Reese.



ATSS TA-2 Executive Summary

Lockheed Space Operations Company

May 1993

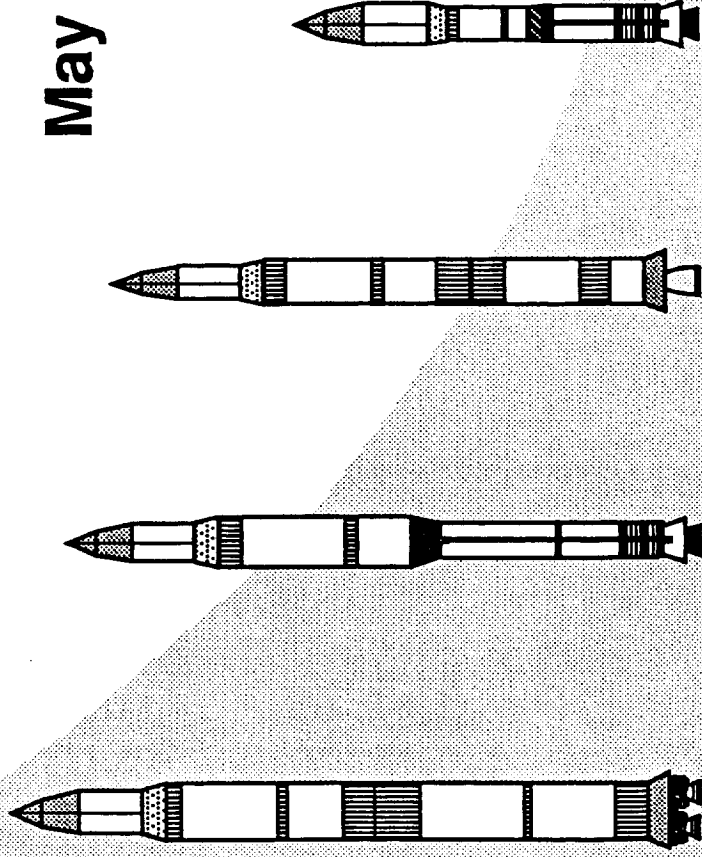




Table of Contents

- **VHM Requirements Definition**
- **Integrated Logistics Support Plan (ILSP)**
- **ATSS Operations Index Evaluation for 50K Launch Vehicles**
- **Lunar HLLV Operations Assessment (RD-170 vs. F-1A)**
- **Launch Site Assessment for Lunar HLLV Dual Launch Concept**
- **Access to Space Option 2- Ground Operations Assessment**
- **CAD Support**
- **Concurrent Engineering Support**
- **Other LSOC Accomplishments**

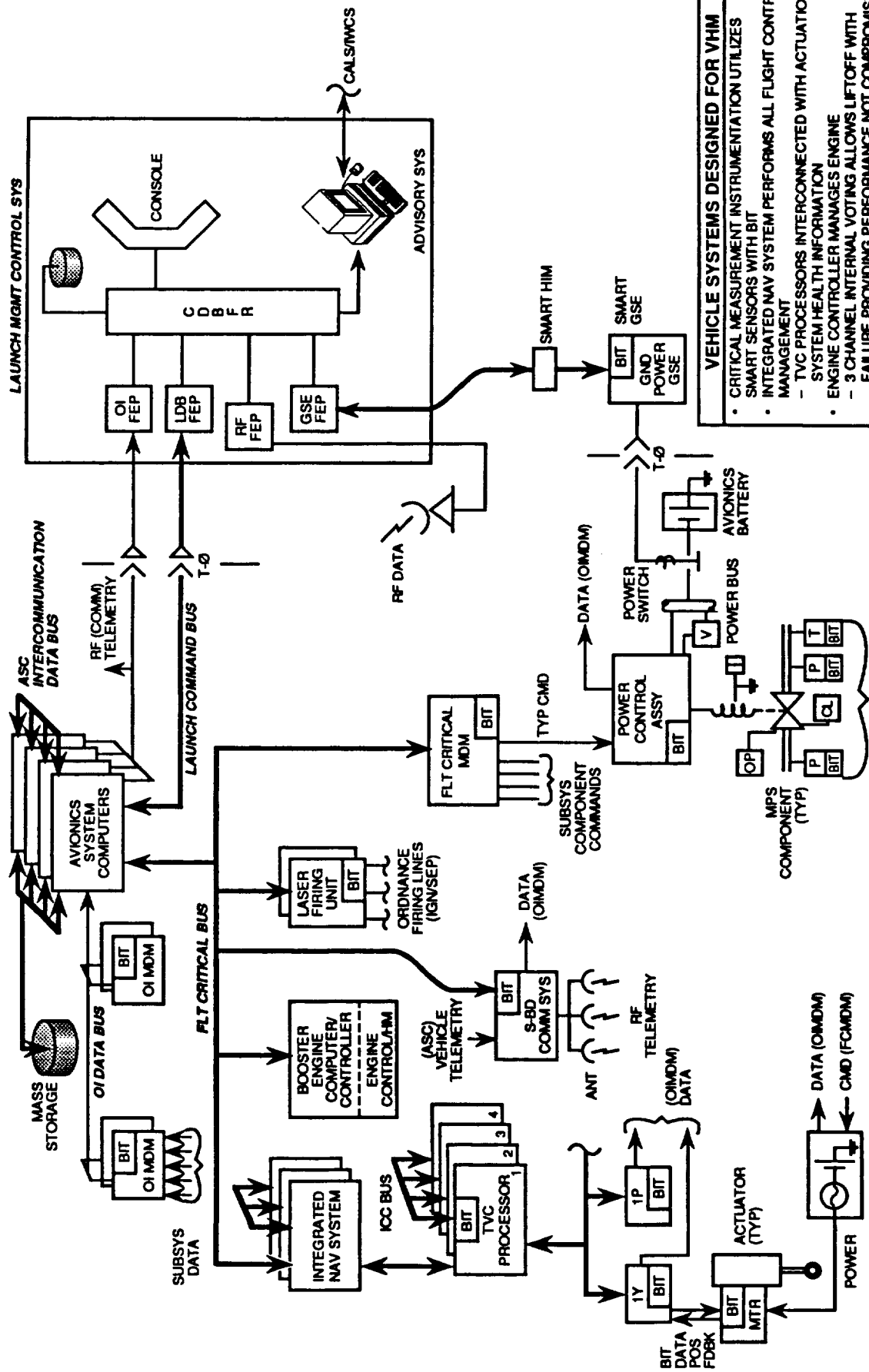


Table of Contents

Lockheed Space Operations Company (LSOC) provided technical support to Lockheed Missiles and Space Company (LMSC) via an Intercompany Work Transfer (IWT) agreement for the ATSS TA-02 basic contract period. LSOC IWT funding was authorized from June 16, 1992 to May 14, 1993. LSOC study activity was managed by the LMSC Principal Investigator and the LSOC task leader, and closely coordinated with the MSFC Contract Technical Representative (COTR) and the KSC Future Launch Systems office. LSOC's primary study focus was in the area of launch and recovery operations, Task 3, with secondary technical support provided to the remaining ATSS TA-02 tasks.

This Interim Report documents in executive summary format the basic contract period study activities. LSOC contributions to the ATSS study are presented in the facing Table of Contents. Complete documentation of each study product/report is available upon request, from the LMSC Study Manager.

VHM Requirements Definition



VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

VEHICLE SYSTEMS DESIGNED FOR VHM

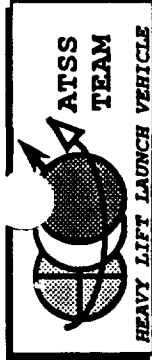
- CRITICAL MEASUREMENT INSTRUMENTATION UTILIZES SMART SENSORS WITH BIT
- INTEGRATED NAV SYSTEM PERFORMS ALL FLIGHT CONTROL MANAGEMENT
 - TVC PROCESSORS INTERCONNECTED WITH ACTUATION SYSTEM HEALTH INFORMATION
- ENGINE CONTROLLER MANAGES ENGINE
 - 3 CHANNEL INTERVAL VOTING ALLOWS LIFTOFF WITH FAILURE PROVIDING PERFORMANCE NOT COMPROMISED
- ASC PERFORMS SYSTEM MANAGEMENT
 - INTERFACE TO GND CONTROL (LMCS)
 - SUBSYSTEMS OPERATIONS AND CONTROL
 - VHM BASED ON SENSOR, SUBSYSTEM PERFORMANCE, BIT, TREND DATA AND SOFTWARE



VHM Requirements Definition

This 60 page ATSS product describes vehicle and ground vehicle health management (VHM) requirements for next generation launch systems. The requirements were then defined for a "generic" liquid propellant medium-lift launch vehicle. The report contents were derived from LSOC experience in Space Shuttle ground processing, launch operations and NLS VHM requirements studies, Lockheed Sanders electronics/fault diagnostics hardware experience, and LMSC fleet ballistic missile systems integration experience.

VHM requirements were divided into the following four categories and further analyzed: 1) methodology; 2) vehicle management; 3) ground management, and; 4) information systems. These VHM categories were described through the phases of engineering development, component manufacturing and acceptance testing, vehicle manufacture/buildup and acceptance testing, launch site integration and launch commit, and mission/post-mission operations.



Integrated Logistics Support Plan (ILSP)

•Logistics Support System

- Operational Logistics Support Plans
- Supportability Effectiveness System
- Logistics Information Management System
- Logistics Management Responsibility Transfer

•Maintenance

- Maintenance Concepts
- Logistics Engineering Analysis
- On line Maintenance
- Off line Maintenance
- Maintenance Verification

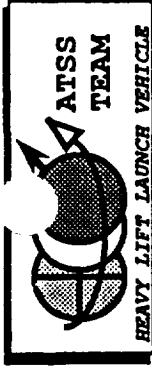
•Supply Support

- Sparing Concepts
- Inventory Management
- Commodities

•Transportation/Handling/Storage

•Training





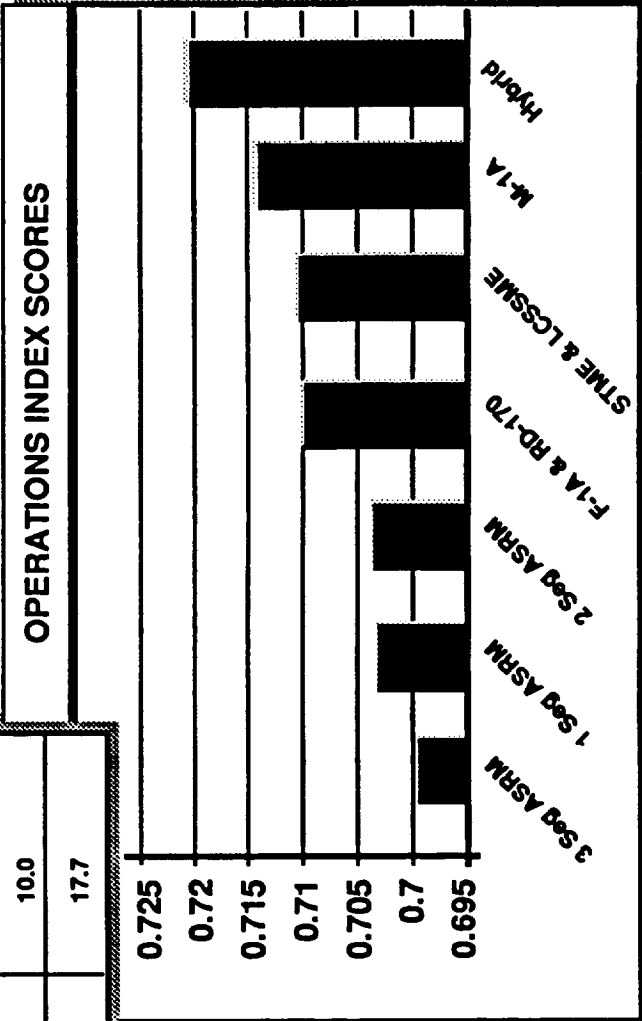
Integrated Logistics Support Plan (ILSP)

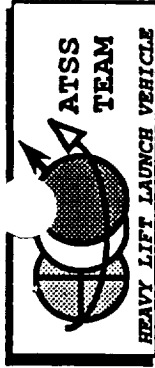
It is NASA policy to address the need for system maintenance or servicing during program formulation. In response to this policy, a preliminary ATSS Integrated Logistics Support Plan (ILSP) was developed. The ATSS ILSP is a comprehensive top-level document, and was prepared as a guide for the further development of a logistics support capability for the proposed ATSS launch system concepts. This document describes the requirements for an ATSS generic integrated logistics support system and establishes the basic disciplines applicable to an operational logistics infrastructure. Included in the preliminary ATSS ILSP is the introduction of a top-level maintenance concept, a sparing and provisioning concept, and a recommendation for a planned logistics management responsibility transfer.



ATSS Operations Index Evaluation (50K Launch Vehicles)

OPERATIONS ANALYSIS METHODOLOGY			
COMPLEXITY FACTOR	LEVEL OF ASSESSMENT		
	SYSTEM	ELEMENT	WEIGHTING (%)
NUMBER OF ELEMENTS	✓		15.6
MANNED/UNMANNED RATING	✓		16.7
PROCESSING CONCEPT	✓		8.9
NUMBER OF FLUIDS	✓		13.3
RELIABILITY	✓		5.6
EXPENDABLE/RECOVERABLE		✓	12.3
PROPELLANT TYPE		✓	10.0
NUMBER OF SIGNIFICANT COMPONENTS		✓	17.7



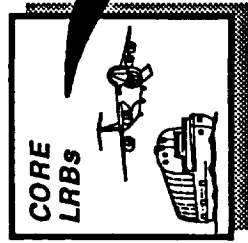


ATSS Operations Index Evaluation (50K Launch Vehicles)

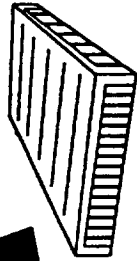
The proposed thirty five (35) ATSS TA-02 50K launch vehicle design configurations were evaluated for operability utilizing an LSOC-developed Ground Operations Index model. This computer aided decision support tool is a customized version of the Manned Transportation System (MTS) Architecture Evaluation Tool (AET) ground operations module. This model provides an operability Figure of Merit (FOM) for each launch system under consideration, and is utilized for relative comparison only. The operability FOM is the weighted sum of a series of operations complexity factor utility values. One of the unique features of this model is that it is useful in the comparative evaluation of launch system designs, where the availability of configuration data is very limited.

The results of our evaluation are seen here. Analysis of the 35 launch vehicle design options categorized each into one of seven configuration common families. The Hybrid/Liquid family scored the highest, followed by the M1-A Liquid/Liquid family. The narrow Index value variation was due to a common second stage and payload carrier design for all 35 launch vehicle options. The primary configuration discriminator is the first and second stage engine selection. Engine operability is not included as a complexity factor input in the current version of our model.

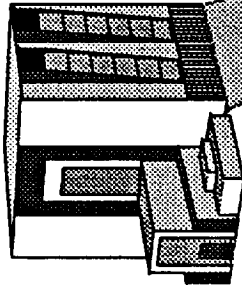
Lunar HLLV Operations Assessment- (RD-170 vs F-1A Engines)



HORIZONTAL PROCESSING FACILITY (NEW)



VEHICLE ASSEMBLY BUILDING (HB-2 OR 4 MODIFIED)



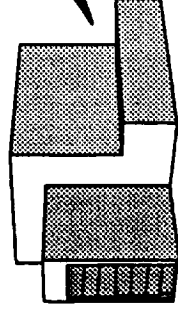
LCC/LPS (MODIFIED)



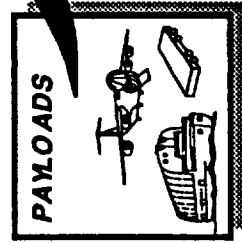
MLT (2 NEW)



PAYLOAD PROCESSING AND ENCAPSULATION FACILITY (NEW)



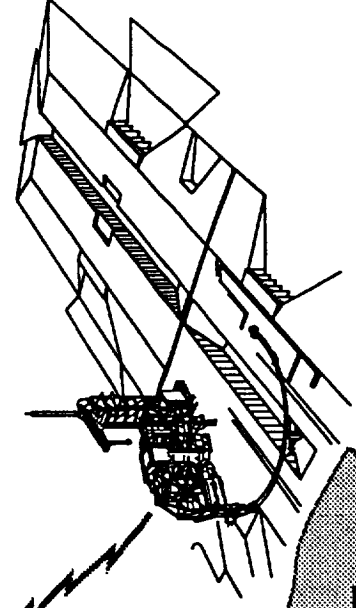
PAYLOADS



PAYLOAD TRANSPORTER (VERTICAL)



3-MILE CRAWLERWAY



PAD 39 A or B w/ EXTENSION

LUNAR HLLV LAUNCH SITE PROCESSING SCENARIO



Lunar HLLV Operations Assessment-- (RD-170 vs F-1A Engines)

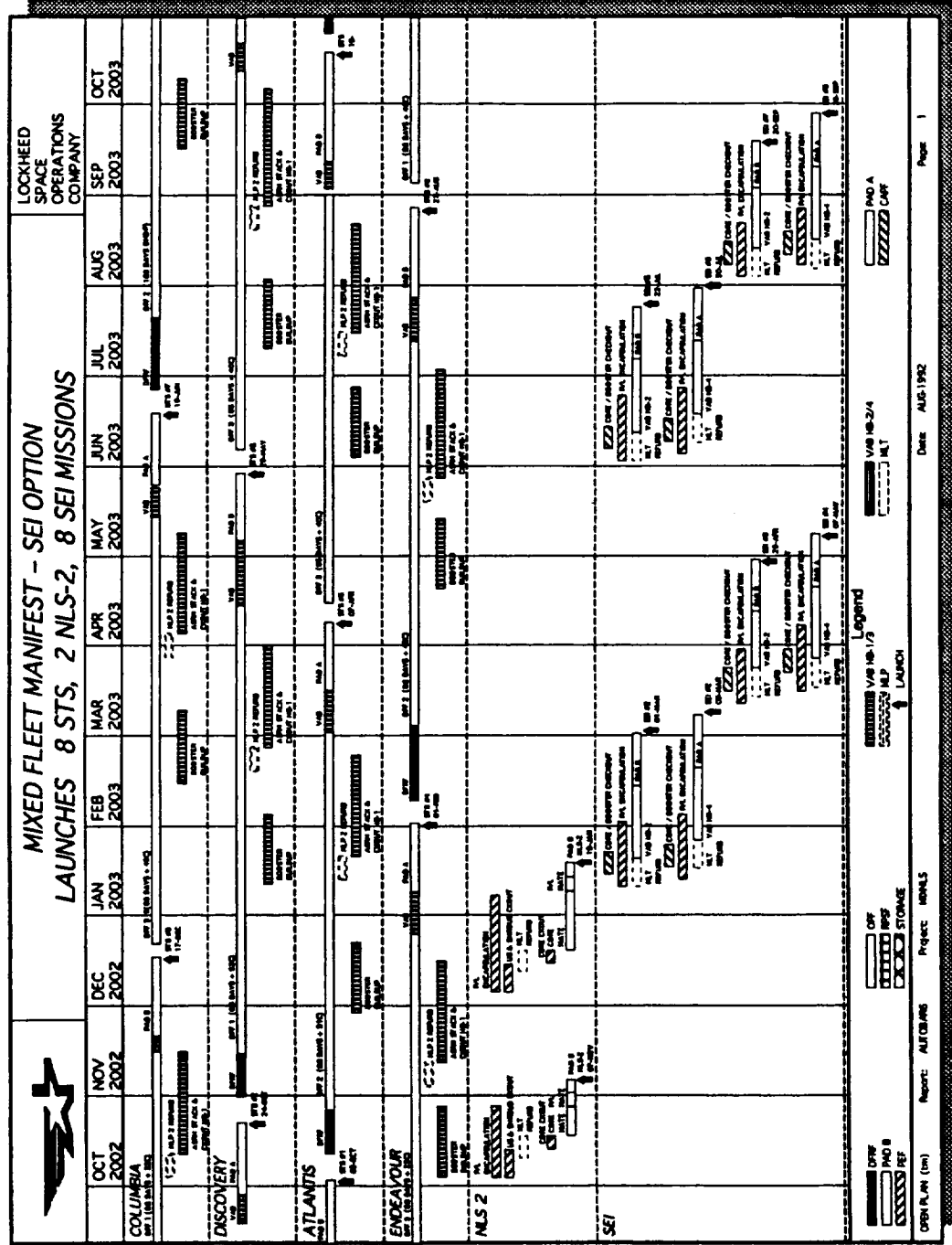
A technical investigation of a proposed lunar HLLV case study was completed for the KSC Future Launch Systems office. This study activity focused on the ground operations assessment of two alternate lunar HLLV configurations, both utilizing the lunar single launch concept. One configuration featured an ET-derived core (SSME engines) with seven LOX/RP-1 strap-on boosters and RD-170 engines. The other configuration utilized the same core stage with eight LOX/RP-1 strap-on boosters with F-1A engines.

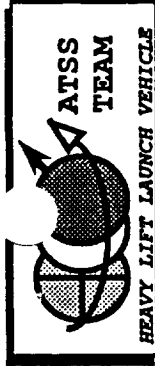
Results of this assessment indicate that there is no significant ground operations discriminator between the two proposed lunar HLLV configurations. The launch site processing scenario, shown here, is interchangeable between the two vehicle options. The predicted scheduled event burden is similar and the launch site station set (facility) solutions are identical. Both options satisfied the minimum lunar launch interval and launch manifest requirements.



Launch Site Assessment - Lunar Dual Launch Concept

First Lunar Outpost - Mixed Fleet Schedule





Launch Site Assessment – Lunar Dual Launch Concept

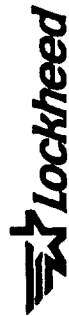
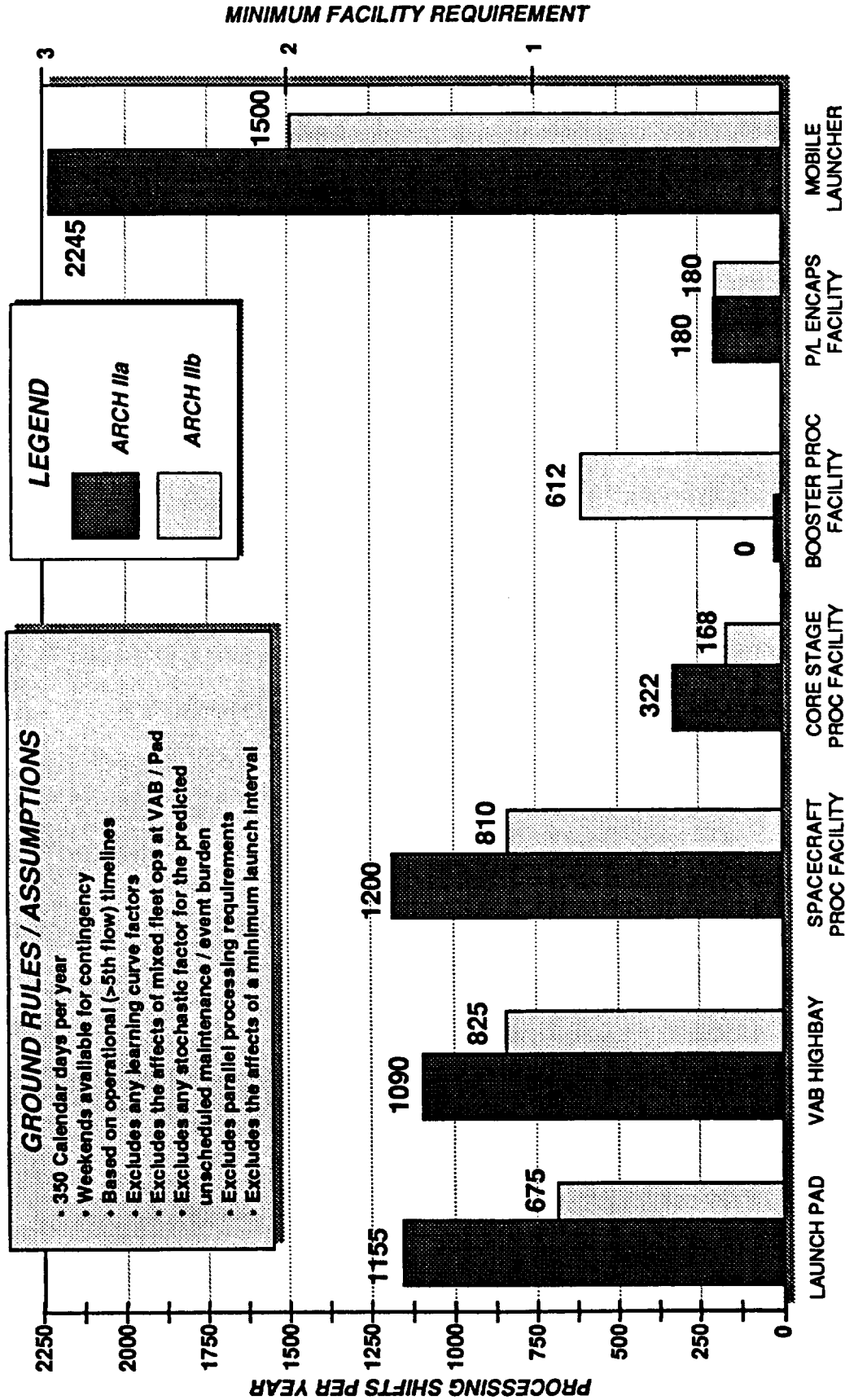
A detailed assessment of the launch site operational impacts for the proposed Lunar dual launch concept was completed at the request of the KSC Future Launch Systems office. The Lunar dual launch concept required two successful ETO launches within an 8 day maximum launch interval for each Lunar cargo or piloted mission opportunity. The nominal interval between Lunar cargo and piloted missions was 60 days. The selected ETO launch system under consideration was an NLS-derived core with 4 STME engines, 2 LOX/RP-1 strap-on boosters with 2 F-1A engines each, and a cryogenic TLI stage for the first launch or a hypergolic kickstage for the second launch in the two launch per Lunar mission sequence.

The issue of launch site schedule feasibility was addressed under a mixed fleet manifest scenario of 8 STS, 8 Lunar and 2 NLS-2 flights annually. A preliminary ground processing scenario was developed, and bottoms-up processing timelines were estimated for each major flight hardware component. These timeline estimates, associated facility resources, and integrated ground processing logic were incorporated into a network-based project management system. The summary output of this effort is shown here in graphic format. The Lunar dual launch concept launch interval requirements were prioritized and maintained. The Lunar, STS and NLS annual flight rates were achieved.



Access to Space Option 2 - Ground Operations Assessment

KSC FACILITY UTILIZATION (5 workdays/week, 3 shifts/day)





Access to Space Option 2 -- Ground Operations Assessment

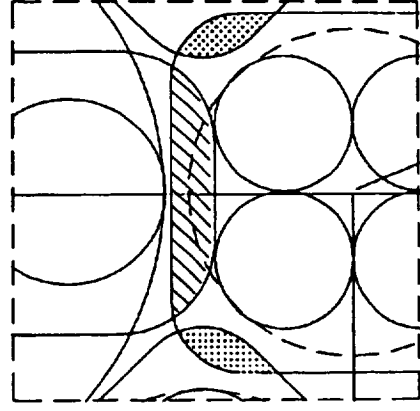
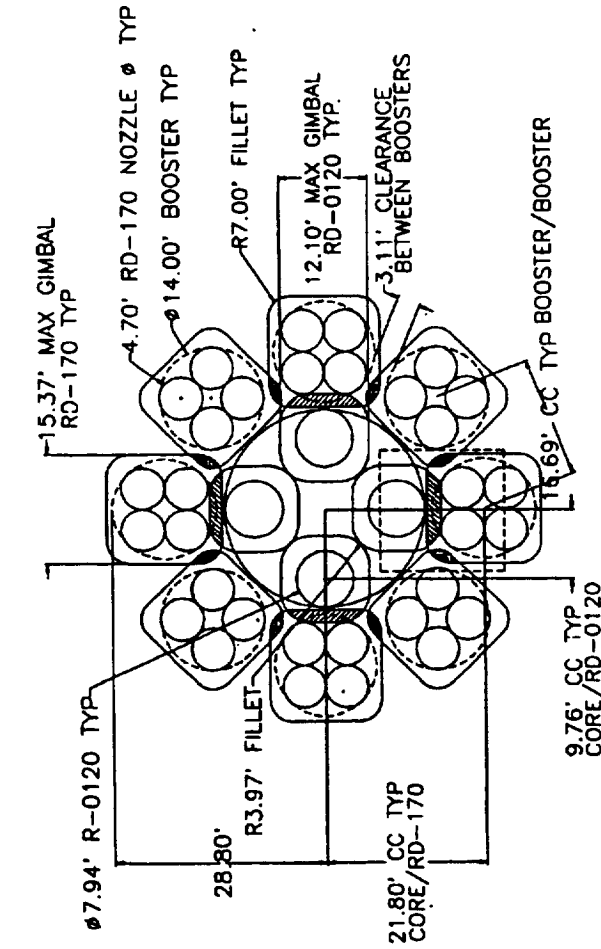
At the request of the KSC Future Launch Systems office (KSC-DE-AST), a schedule sensitivity trade study was completed for the Access to Space Option 2 Launch Vehicle Team. This trade study focused on the launch vehicle Reference Family 1b Architecture 1la and 1lb options. All conceptual vehicle configurations in the subject architectures were variations of a 1.5 stage core with four (4) STME engines. Architecture 1la featured the HL-42 spacecraft launched at a maximum annual flight rate of 3 for the manned configuration and 17 for the unmanned configuration. Architecture 1lb featured the Crew Logistics Vehicle with pressurized payload (CLV-P) launched at a maximum annual flight rate of 3 for the manned configuration and 6 for the unmanned configuration. Both architecture options included an unmanned 50K launch system with a maximum annual flight rate of 3.

Ground processing timelines were developed for each launch system based on analogous data from similar advanced program studies. NLS-2 data was utilized for the 1.5 stage core, cryogenic upper stage and payload encapsulation requirements. PLS HL-20 and CLV-P data were utilized for the HL-42 and CLV spacecraft, respectively. Hybrid booster fuel segment processing was assumed to be analogous to the RSRB aft segment/skirt functions at the launch site, and the hybrid fuel and oxidizer "segment" mating on the mobile launcher was assumed to be analogous to RSRB aft and forward segment stacking and mating operations.

The results of this assessment are depicted in the facing chart. Architecture 1la has a greater scheduled event burden (30%) and requires a significantly higher DDT&E Investment than Architecture 1lb (i.e., more facilities, activation, and launch operations resources). Of the two architectures, 1lb provides a higher degree of schedule flexibility (and thus the capability to manage time), and offers the better opportunity to optimize the launch site Shuttle transition requirements.



CAD Support



NOTES:

- ±8' GIMBAL - ALL ENGINES
- 1.00' OFFSET BETWEEN CORE & BOOSTERS

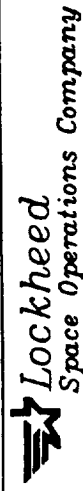
NO RD-0120/RD-0120 GIMBAL OVERLAP

⊙ TOTAL RD-170/RD-170 GIMBAL OVERLAP = 37.20 SQ. FT.

⊙ TOTAL RD-0120/RD-170 GIMBAL OVERLAP = 56.12 SQ. FT.

TOTAL GIMBAL OVERLAP AREA = 93.32 SQ. FT.

CONFIGURATION 2B OPTION 1
EHLLV CORE W/4 RD-0120s & 8 RD-170s



DRAWN BY ARLENE REESE	ADVANCED PROGRAMS STUDY TEAM
DATE 2-5-93	SIZE A
DWG NO. 28-4	REV



CAD Support

A package of HLLV first stage engine and gimbal arrangements was developed. The 40 page document was developed using AutoCad and contains drawings of alternate engine and gimbal arrangements for 13 candidate Lockheed lunar HLLV concepts. The HLLV concepts were grouped into parallel and series burn vehicle configurations. The drawings presented alternate methods of arranging the HLLV engines and actuators to minimize both total boattail area and engine gimbal overlap (assuming 5-8 degree gimballing). Booster-to-booster and booster-to-core clearances were also calculated in the drawings to address the concerns of launch vehicle accessibility during preflight assembly.

CAD drawings were also completed for two alternate propellant feedline systems for a candidate Lockheed 50Klb vehicle concept using a LOX/RP-1 fueled F-1A engine first stage and LOX/LH2 fueled SSME engine second stage. The A and D-size drawings were delivered to the ATSS study manager and MSFC COTR.



ATSS Workshop – Concurrent Engineering

Operations Impacts on HLLV Design Characteristics



ATSS – TA-2 Concurrent Engineering Meeting 8/5/92

HLLV LAUNCH SYSTEM "IDEAL" DESIGN CHARACTERISTICS

MINIMIZE VEHICLE VOLUME

- Minimum Vehicle Height and Aspect Ratios
- ☐ All Stages Use Multi-cell Propellant Tanks (Shorter Length)
- ☐ All Stages Use Toroidal Air Propellant Tanks (Shorter Length)
- ☐ All Stages Use Common Bulkheads
- ☐ Minimize Dry Weight of Each Stage
- Minimize Total Vehicle Dry Weight
- ☐ Nuclear Upper Stage Propulsion System (Reduce Volumetric Requirements and Weight)
- ☐ Utilize NASP Technology in Upper Stages (i.e. Slush Hydrogen)
- ☐ Utilize High Density Propellants
- ☐ Use High Density-Impulse Propellants for All Stages

MINIMIZE NUMBER OF ENGINES

- Minimum Number of Engines
- Minimum Number of Engines per Stage (For Many Reasons)
- Minimum Number of Different Engines



ATSS – TA-2 Concurrent Engineering Meeting 8/5/92

HLLV LAUNCH SYSTEM "IDEAL" DESIGN CHARACTERISTICS

MINIMIZE SAFETY HAZARDS

- Minimize Range Safety Hazards (Tracking, Destruct, Disposal)
- No Nukes
- Minimize Blast Danger / Quantity Distance
- Minimize Number of Hypersonic Operations



ATSS – TA-2 Concurrent Engineering Meeting 8/5/92

HLLV LAUNCH SYSTEM "IDEAL" DESIGN CHARACTERISTICS

MINIMIZE NUMBER OF FLIGHT ELEMENTS

- Minimum Number of Elements (Parts Count)
- ☐ Minimize Number of Vehicle Stages / Boosters
- ☐ Minimize Number of Stages to Reduce Number of Contractors and Improve Overall Reliability
- ☐ Minimize Staging
- ☐ Many Stage Vehicle
- Minimize Number of Reusable Flight Elements
- Minimize Number of Solid Segments
- Minimum Number of Different Propellants
- ☐ Launch Stack with Fewest Elements

MAXIMIZE FLIGHT ELEMENT / COMPONENT COMMONALITY

- Minimize Number of Different Flight Element Diameters
- ☐ Common Core Vehicle Configuration for Lunar and Mars Missions
- ☐ Monolithic Vehicle for Lunar Mission (Add Boosters for Mars Mission)
- ☐ Vehicle Designs with Minimum Impact from Using Alternative Elements (Engines, Boosters)
- Minimum Number of Different Engines
- ☐ Shortest Possible Development Time (Highest Inheritance)

Operations Impact





ATSS Workshop – Concurrent Engineering

A concurrent engineering meeting was held with all the ATSS TA-2 study team members to formulate the desirable characteristics for comparing HLLV candidate configurations. Many of these characteristics were partially driven by the desire to have an operable launch system. The characteristics impacted by operations include minimizing environmental impacts, design complexity, vehicle volume, number of flight elements, number of engines, and safety hazards. The characteristics impacted also include maximizing existing infrastructure, launch availability, operability, and flight element commonality.

Three categories were defined for the HLLV concepts, based on cost and risk. The categories, minimum up-front cost, minimum recurring cost, and minimum risk, each emphasized different design characteristics.

Each of the HLLV design characteristics defined at the concurrent engineering meeting were derived using the QFD affinity process. Some of the operations affinity inputs include minimizing vehicle height, weight, and aspect ratio, minimizing the number and type of engines and propellants on each stage, minimizing the number of solid segments, hypersonic operations, flight element diameters, and refurbished/reused flight elements.



Other LSOC Accomplishments

- FLO Operations Concept Document
- FLO Landing and Recovery Concepts
- Lunar HLLV Mixed Fleet Assessments
- ATSS Operations Concept Document (Outline)
- HLLV Technical Interchange Meetings (TIMs)
- ATSS Workshop - Launch System Concepts
- Vehicle Health Management Workshops



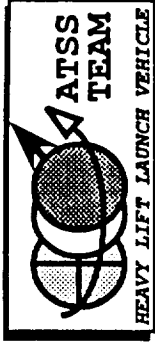
Other LSOC Accomplishments

A number of miscellaneous tasks were completed during the ATSS TA-02 basic contract period at the request of and in support to the LMSC Study Manager, the MSFC Technical Monitor and the KSC Future Launch Systems office. The significant tasks are shown on this chart. A detailed redline/review of the *First Lunar Outpost (FLO) Operations Concept Document* and the *FLO Landing and Recovery Concepts Document* was performed. A launch site resource and schedule optimization trade was completed for a series of Lunar HLLV mixed fleet manifest scenarios. An ATSS HLLV Operations Concept Document outline was formulated. HLLV Technical Interchange Meetings at KSC, Stennis Space Center (SSC), Lockheed Sunnyvale and LaRC were supported.

A number of ATSS Workshops were supported including the Concurrent Engineering, Launch System Concept development and VHM workshops. AutoCad support was provided for launch vehicle/booster and engine CAD layouts as the preferred ATSS launch system concepts matured. Lunar HLLV design requirements and guidelines were formulated using the *First Lunar Outpost Requirements and Guidelines (FLOG)* document, dated June 10, 1992. The requirements or guidelines/derived requirements which impact operations were listed.

11.0 Health Management System Definition Study

This section contains a copy of a vehicle-independent, launch system health management requirements assessment that was performed by Mr. Steve Black of LSOC, under an intercompany work transfer for LMSC. The purpose of the assessment was to define both health management requirements and the associated interfaces between a generic advanced transportation system launch vehicle and all related elements of the entire transportation system, including the ground segment.



1

HEALTH MANAGEMENT SYSTEM DEFINITION STUDY

STEVE BLACK
LSOC

PROJECT ENGINEERING AND TEST INTEGRATION
ADVANCED PROGRAMS



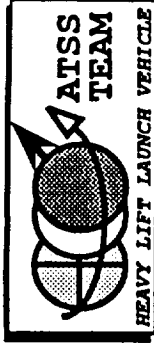


TABLE OF CONTENTS

INTRODUCTION

DEFINITION

HEALTH MANAGEMENT SYSTEM CATEGORIES

- **METHODOLOGY**
- **VEHICLE MANAGEMENT**
- **GROUND MANAGEMENT**
- **INFORMATION SYSTEMS**

INTEGRATED HMS FOR ATSS MEDIUM LIFT LAUNCH VEHICLE

- **PROCESSING SCENARIO**
- **VEHICLE DESIGN**
- **FULLY INTEGRATED HMS ARCHITECTURE**
- **VHM SYSTEM ARCHITECTURE**
- **LAUNCH BOOSTER MANUFACTURING PROCESS**
- **LAUNCH SITE OPERATIONS**
 - **BOOSTER**
 - **UPPER STAGE**
 - **CARGO**
 - **INTEGRATED**



INTRODUCTION

The purpose of this presentation is to provide in a useful manner ground and vehicle health management definition for the next generation launch systems based on current program direction, vehicle design concept, available and future ground launch facilities.

The substance of this report were derived from experience in Space Shuttle processing / launch operations and NLS studies by Lockheed Space Operations Company, DoD high technology aerospace electronics / fault diagnostics development by Lockheed Sanders, and ballistic missile weapon systems manufacturing by Lockheed Missiles and Space Co.

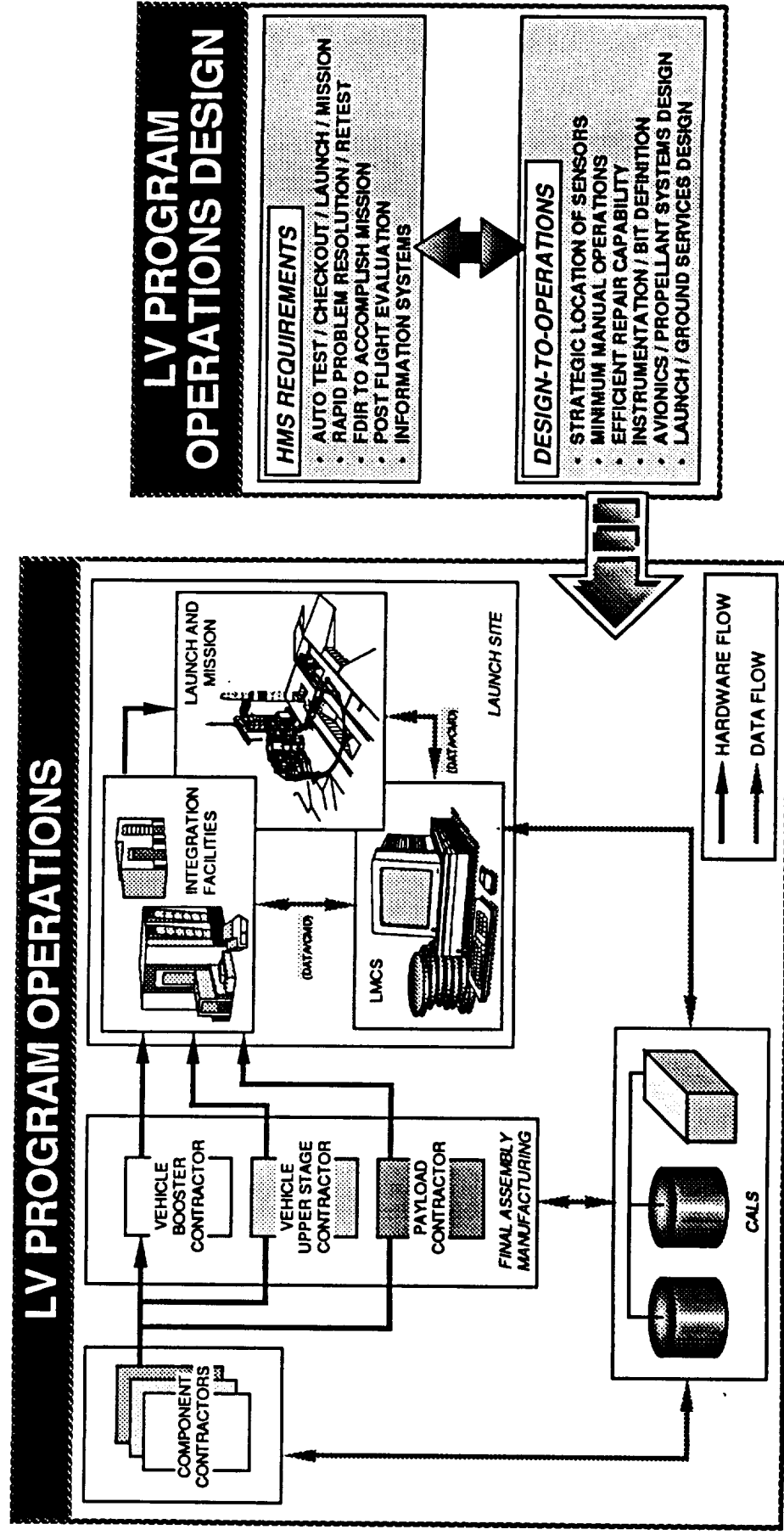
INTRODUCTION (cont'd)

The investment in high technology health management systems for new launch vehicles should be made based on operations optimum expense over the projected program lifetime and specific mission objectives. If for instance, an unique launch vehicle is developed for a purpose requiring say less than 4 launches per year, it would not be advantageous to develop automated VHM systems that cost more to maintain than that of less technology manual systems. But, given a general purpose booster with a high launch rate (say 20 per year) and launch-on-demand requirements, a more elaborate HMS should be employed incorporating artificial intelligence, robotics, and on-board checkout.

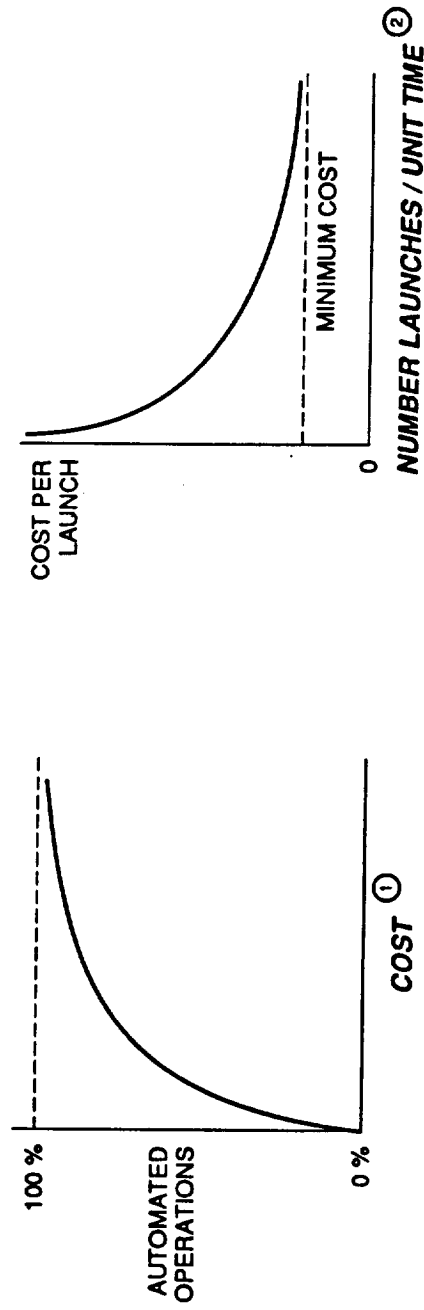
The HMS for a particular launch vehicle evolves with the launch vehicle program development. The optimization process is effective only with a "design to operations" approach where representatives from all aspects of the launch vehicle's life cycle are represented in a continuous improvement TQM environment.

INTEGRATED PROGRAM DEVELOPMENT AND OPERATIONS

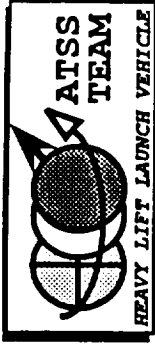
OBJECTIVE: *Efficient, flexible high energy launch vehicle*



LAUNCH VEHICLE OPERATIONS COST PROFILE



- ① COST PROFILE REPRESENTS EXPENSE TO DEVELOP, BUILD AND MAINTAIN AUTOMATED SYSTEMS TO SUPPORT MANUFACTURING AND LAUNCH OPERATIONS.
- ②
 - MINIMUM COST PER LAUNCH IS BASED ON VEHICLE CONSUMABLES, GSE/FACILITY O&M AND PROGRAM SUPPORT O&M.
 - COST PER LAUNCH CURVE AND MINIMUM COST LINE ARE BIASED UP/DOWN BASED ON THE AMOUNT AND COST OF SUPPORTING AUTOMATED OPERATIONS.



DEFINITION



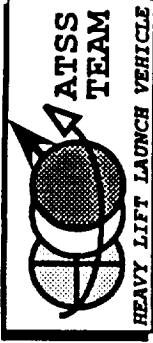
DEFINITION

The Health Management System for ATSS has a broad definition encompassing "cradle to grave" vehicle life cycle data management, test philosophy and maintenance concept. HMS can be grouped into four major categories:

- Methodology
- Vehicle Management
- Ground (including vehicle / ground integrated) Management
- Information Systems

Each category can then be subgrouped and described in phases:

- Engineering development, manufacturing and acceptance (component level)
- Vehicle buildup, manufacture and acceptance (vehicle element)
- Launch site integration and launch commit (preflight verification)
- Mission / post-mission (flight operations)



DEFINITION (CONT'D)

The ultimate objective of the HMS is to minimize operational costs while maximizing launch / mission reliability.

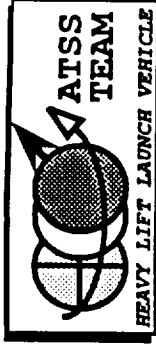
❑ Manufacture / Launch Process

- Expedite Test and Checkout
 - Automated Systems
- Quick Problem Resolution
 - Rapid Fault Isolation / System Safing
 - Minimize Vehicle Damage
 - Minimize Troubleshooting
 - Efficient Repair
 - Rapid Retest and Flight Readiness Re-verification

❑ Mission Operations

- FDIR to Accomplish Mission
- Downlink Data Evaluation

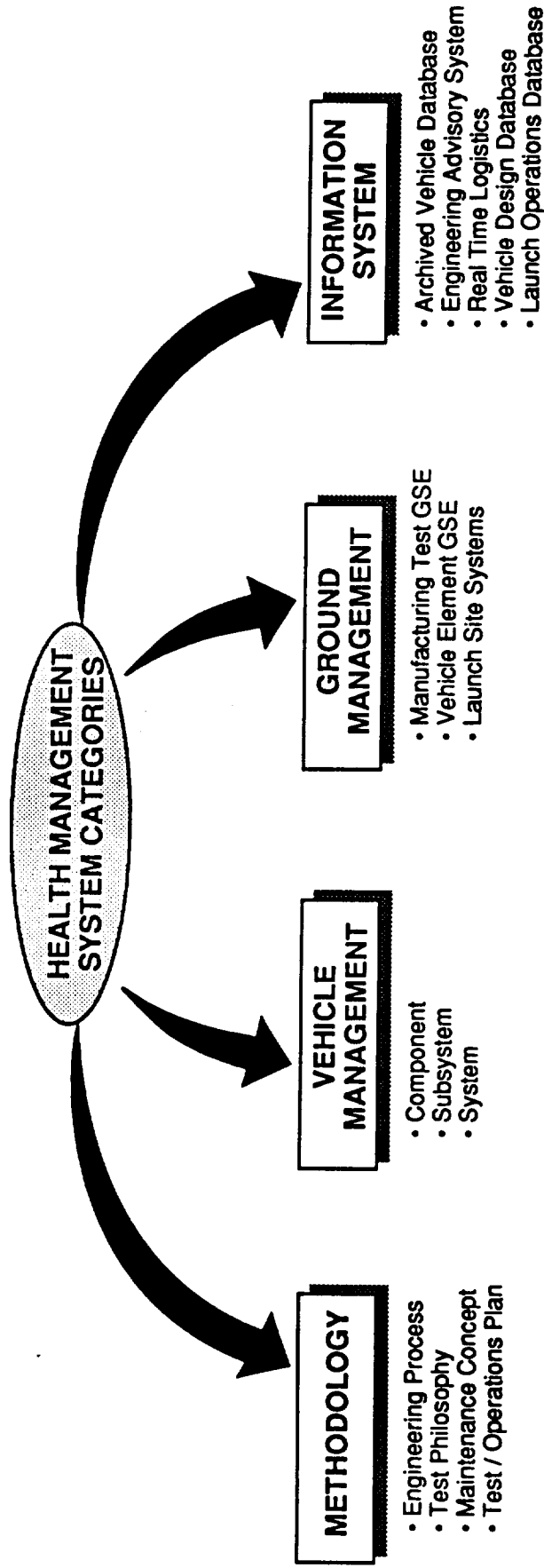
Vehicle Health Management (VHM) is then a subset of the Health Management System and is best described as a system using BIT, instrumentation, ground and vehicle software with operator interfaces that vary according to the amount of designed-in human intervention, and integrated with a comprehensive information data system.



HEALTH MANAGEMENT SYSTEM CATEGORIES

ATSS Vehicle Health Management System Definition Study

HMS CATEGORIES — OVERVIEW



GOAL: Define Unique VHM Requirements for Each Phase of Vehicle Life Cycle from Engineering Development Through Post-Mission Analysis.



11

HMS CATEGORIES **METHODOLOGY**

HMS CATEGORIES — METHODOLOGY

12

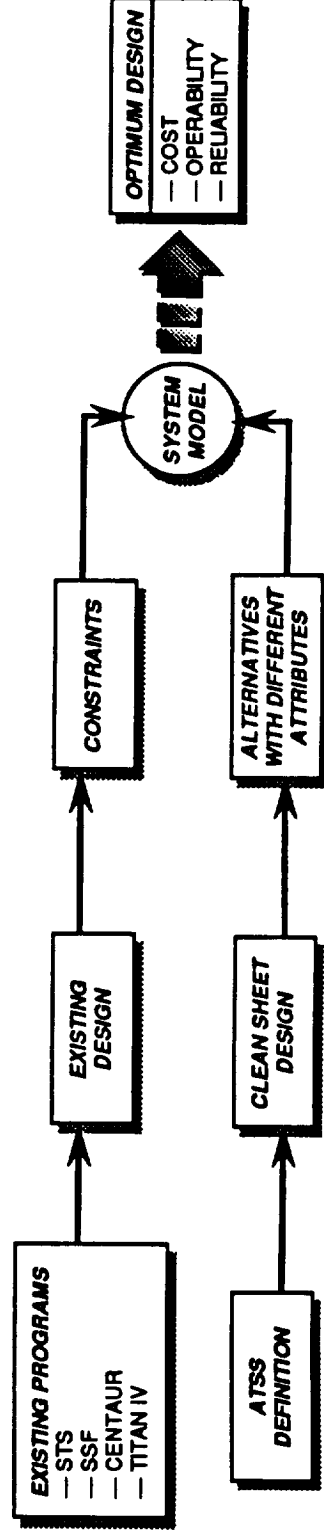
ENGINEERING PROCESS

□ Design to Operations

- Using Program Model / Definition (Number of Launch Vehicles per Year, Surge Requirements, Launch on Demand, Mission Models), Develop Facility, Vehicle Manufacturing and GSE Requirements
- Develop Operations Model Based on System Design Characteristics, Cost and Previous Program Experience
- Ensure all Phases of the Program are Developed for Operability Goals

- Requirements
- Designs
- Trades
- Manufacturing
- Processing

- Use CIP / TQM Process Involving Design, Manufacturing and Launch Operations Personnel



HMS CATEGORIES —METHODOLOGY

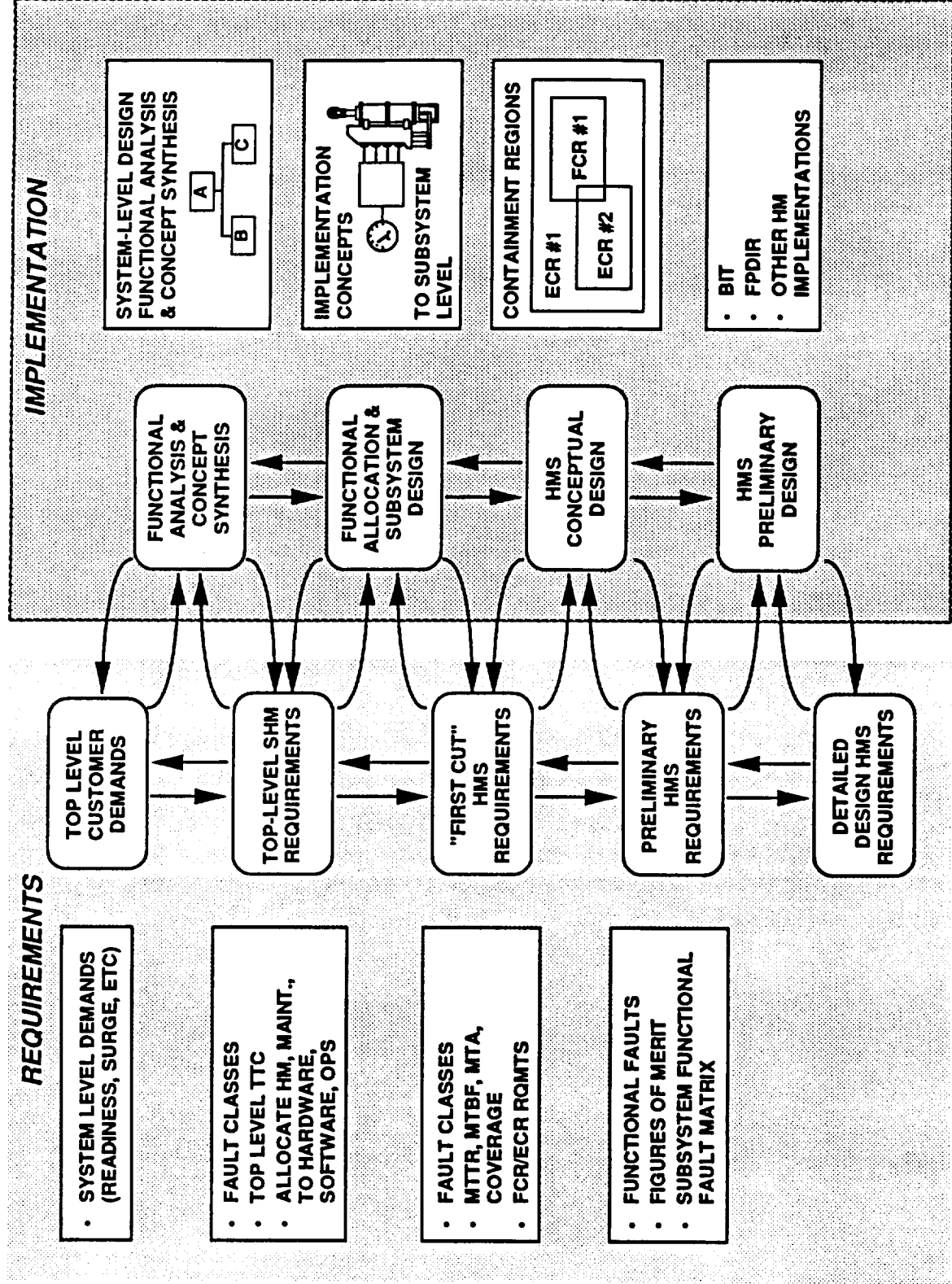
13

ENGINEERING PROCESS

- ❑ HMS Systems Design Process
 - Design Requirements and Implementation Definition
 - System / Program Requirements
 - Subsystem Requirements, Limitations and Fault Classes
 - Component Requirements, Limitations and Functional Faults
 - Technology Applications
 - Proper Technology Applications to Yield Optimum Fault Diagnostic Capabilities, Program Objectives and Costs

HMS SYSTEMS DESIGN PROCESS

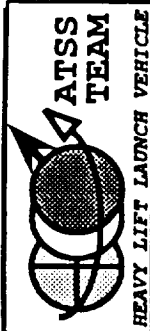
14



ATSS Vehicle Health Management System Definition Study

HMS SYSTEMS DESIGN PROCESS

	INITIAL REQUIREMENTS	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	DETAIL DESIGN	FABRICATION & TEST	DEPLOYMENT & OPERATIONS
QUANTITATIVE REQUIREMENTS	• Availability • Reliability Allocation • Margin Philosophy	• Maintainability Metrics, MTTR, etc. • Time to Criticality at System Level • Design Margin Establishment	• Dependability Metrics Refinement • Time to Criticality at Subsystem Level • Margin Requirements	• Final Margin & Reliability Requirements	• Requirements Updates	• Requirements Updates
QUALITATIVE REQUIREMENTS	• System Fault Tolerance Rqmts. • System Fault Allocation Rqmts. • Fault Classes for Fault Tolerance • Isolateability	• Subsystem Fault Tolerance Rqmts • Subsystem Functional Fault Rqmts.	• Fault Injection Rqmts. • Software and Hardware Operational Rqmts.	• Final System & Subsystem Component Requirements	• Requirements Updates	• Requirements Updates
FAULT SET DEFINITION	• Fault Classes • Major Implementation Fault Types (Engine Out, Electronics, Etc.)	• Subsystem Functional Faults (Top Down) • Preliminary FMEA (Bottoms Up)	• Refinement of FMEA (Quantitative Failure Rates) • Gathered Fault Combination Method (GFCM) — Fault Set Reduction for Fault Injection	• Final FMEA • Refine Fault Set Reduction for Fault Injection	• Updates to Fault Set	• Updates to Fault Set



ATSS Vehicle Health Management System Definition Study

HMS SYSTEMS DESIGN PROCESS

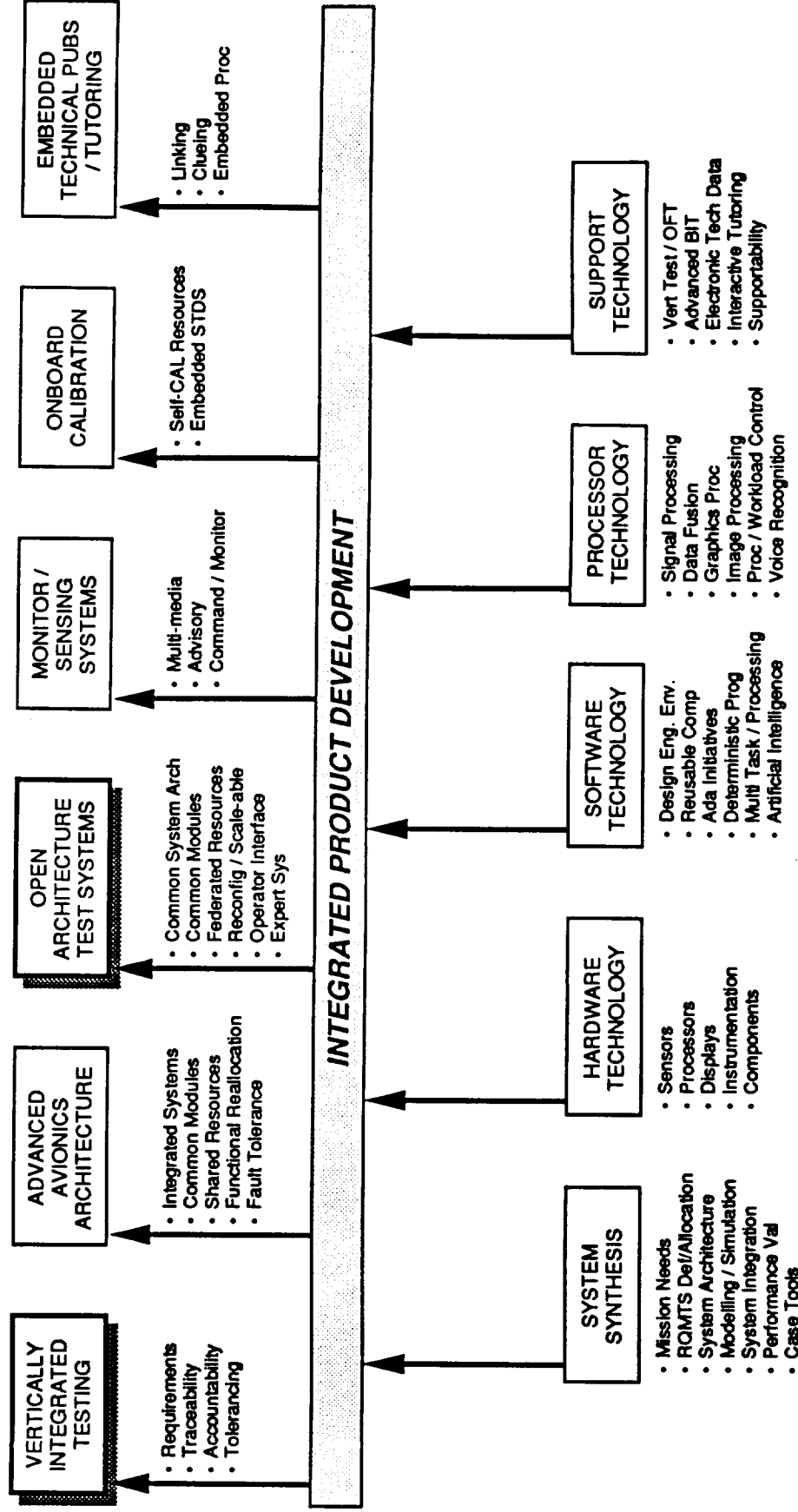
16

	INITIAL REQUIREMENTS	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	DETAIL DESIGN	FABRICATION & TEST	DEPLOYMENT & OPERATIONS
FAULT ANALYSIS & MODELING	System Cost and Reliability Trades	- System Interaction Time to Criticality - Functional Fault Matrix - Initial Behavioral Model	- False Alarm Analysis - Detailed System Modeling - Detailed Reliability Modeling - Simulation with Fault Injection - Cost / Reliability Analysis for Parameters	- Simulation with Fault Injection - Detailed Alarm/Action Threshold Analysis - Detailed False Alarm Analysis	- Fault Injection into As-Built System - Alarm Threshold Testing with Integrated HW/SF - System Characterization - Model Updates	- System Characterization - Model Updates - Fault and Contingency Analysis
SYSTEM DESIGN	- Initial System Concept - Operation & Maintenance Concepts - Health Management - Data Flow Plan	- Initial Subsystem Concept - ECR/FCR Definition at Subsystem Level - First Cut at Parameter Set - Degree of System Autonomy (Human Role) - HM Software Plan Integration - Design Provisions for Passive Fault Tolerance	- Refinement of HM Architecture Concepts - Detailed ECR/FCR - Complete FPDIR Plan - Parameter Refinement, Algorithm Dev. - Sensor Selection	- Design Implementation Detailing - HM Thresholds (Alarm, Reaction, Persistence, etc. Established) - Detailed Data Mgmt. Plan - HM Ground Equip. Design Refinement	Design Feedback Threshold Adjustment from System Characterization	- System Characterization - Design Updates - Contingency Plans
VERIFICATION & VALIDATION	—	- V&V Plan Draft for SHM - Allocation of V&V Methods: Test, Analysis, Proof, Simulation	- Incorporate Prelim. FMEA into V&V - Define Fault Injection Techniques - Proof of Key Algorithms	- Test Procedures - V&V by Analysis, Simulation Test, & Formal Proof	Subsystem and System Testing Under Stressing Conditions & Fault Conditions	Testing Updates

HMS SYSTEMS DESIGN PROCESS — TECHNOLOGY APPLICATIONS

17

BLENDED TECHNOLOGIES YIELD AFFORDABLE AND CAPABLE SYSTEMS SOLUTIONS



TECHNOLOGIES

SOURCE: LOCKHEED SANDERS

HMS CATEGORIES — METHODOLOGY

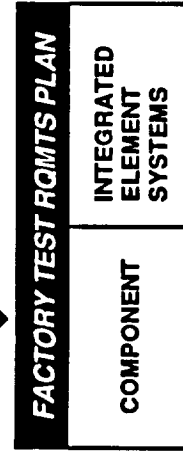
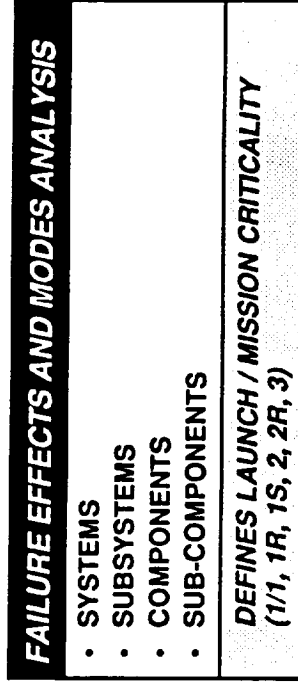
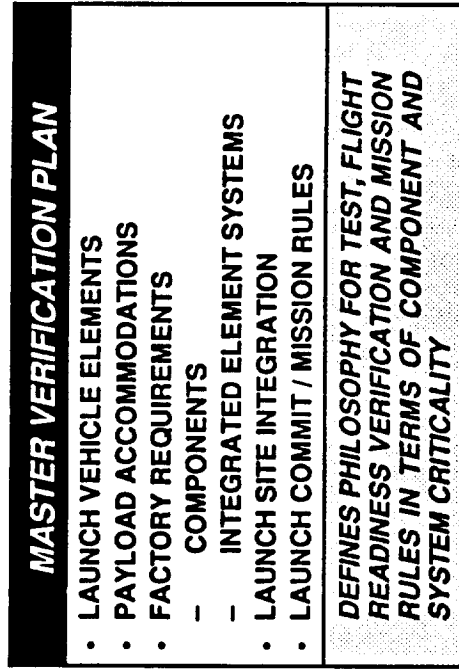
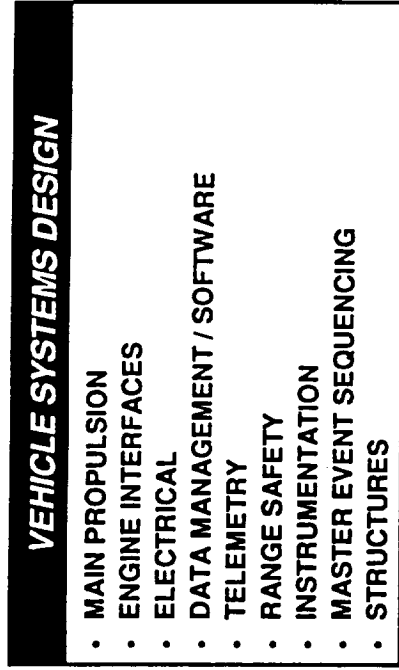
TEST PHILOSOPHY

- ❑ **Manufacture Acceptance Test Requirements**
 - Subcomponent / Component Level Design ATP
 - Internal Redundancy Verification Requirements
 - Flight Environment Requirements
- ❑ **Vehicle Manufacturing and Launch Site Delivery Acceptance Requirements**
 - LRU Receiving Acceptance
 - Subsystem Checkout Requirements
 - Design Verifications
 - Performance Verifications
- ❑ **Preflight Verification (Launch Site)**
 - Subsystem Preflight Service Requirements
 - Element Integration and Interface Verification Requirements
 - Subsystem / System Preflight Verification Requirements
- ❑ **Launch Commit Criteria**
 - Minimum Component / Subsystem Requirements to Reliably Achieve Mission
 - LCC Verification Methodology

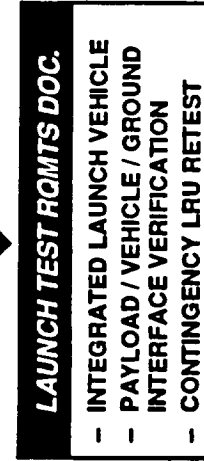
HMS CATEGORIES — METHODOLOGY

19

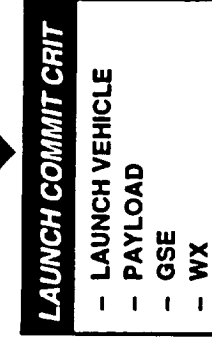
TEST PHILOSOPHY



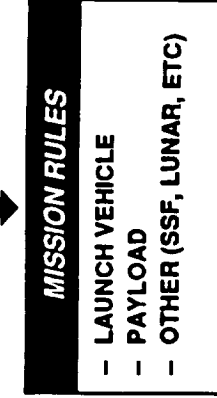
TEST DOCUMENTS
- ATP
- FLT ENVIR TESTS



TEST PROCEDURES SOFTWARE



- LCD CONTINGENCY PROCEDURES
- GLS SOFTWARE
- ONBOARD SOFTWARE



- MISSION CONTROL PROCEDURES
- ONBOARD SOFTWARE

HMS CATEGORIES — METHODOLOGY

MAINTENANCE CONCEPT

- ❑ Logistics / Quality Methodology Utilizing Computer-Aided Acquisition and Logistic Support (CALS)
 - Parts Availability Database
 - Inspection History Database
 - Automated / Paperless System for Parts Transfer
 - Database for Individual Parts from Manufacture Through Mission. Significant Part Information Archived Post-flight (Software System AI-based).
 - CAD Database of Component Information and Vehicle Assembly Drawings

MAINTENANCE CONCEPT (cont'd)

□ MIL-STD-1814 Integrated Diagnostics Standard

- Generic Requirements and Verifications for Incorporating Integrated Diagnostics (ID) into Program Acquisition, Fault Diagnostic Capability and Operations Maintenance**
- An Integrated Approach to Achieving a Balanced Diagnostic Capability**
 - Integration of Embedded, Support Equipment, and Manual Techniques to Provide Complete Coverage of Diagnostic Information Needs**
 - Integration of All Needs for Diagnostic Information to Minimize Overall Diagnostics Required and Optimize Performance**
- The ID Design is Developed from a Systems Engineering Environment (i.e., Design to Operations)**

HMS CATEGORIES — METHODOLOGY

21

MAINTENANCE CONCEPT (cont'd)

- ❑ Problem Disposition / Resolution
 - Problem Reporting System
 - Near Paperless
 - Utilize Information System to the Greatest Extent for Problem Documentation and Database Information
 - Electronic Signature Approval
 - LRU Repair / Replacement
 - Subcomponent (Module) Level Isolation via BIT
 - In-place Repair / Module Replacement if Practical
 - Launch Site Depot Repair
 - Design Review Board
 - Electronic Signature Approval for Sustaining Engineering Changes
 - Sustaining Engineering Changes Processed via CAD
 - Utilize Retina Projection Technology
 - Provides Means for Paperless Work Environment
 - Expands Information Available to Technician Performing Work



HMS CATEGORIES — METHODOLOGY

22

OPERATIONS PLAN

❑ Manufacturing Planning

- Standardize Parts, Procedures, Test and GSE
- Utilize AI Tools for Schedule / Planning
- Standardize Database Information for Components, Subsystems and Systems
- Launch Vehicle Element Delivery Readiness Verification

❑ Launch Site Integration (ITL)

- Transportation Plan (GSE and Vehicle Verification Requirements)
- Standdown / Storage Requirements
- Major Flow Milestone Readiness Verification Requirements
- Launch Site Vehicle Processing Standards
 - Element Hardware Subsystem Test and Service Requirements
 - Element Interface Verification Test Standards
 - Automated Ground Launch System Mission Software Set Generation
- Integrated Vehicle System Launch Readiness Verification Standards
- Launch Commit Criteria Standards

HMS CATEGORIES — METHODOLOGY

OPERATIONS PLAN (cont'd)

❑ Mission Planning

—Prelaunch

- Payload Integration Requirements and Standards (Data, Command and Ground Interface Definition)
- Vehicle Flight Integration Requirements
- Data / Command Requirements

Loads Analysis

Automated Trajectory / Flight Software Generation

Automated / Real Time Wind Load Coefficient Generation

—Mission

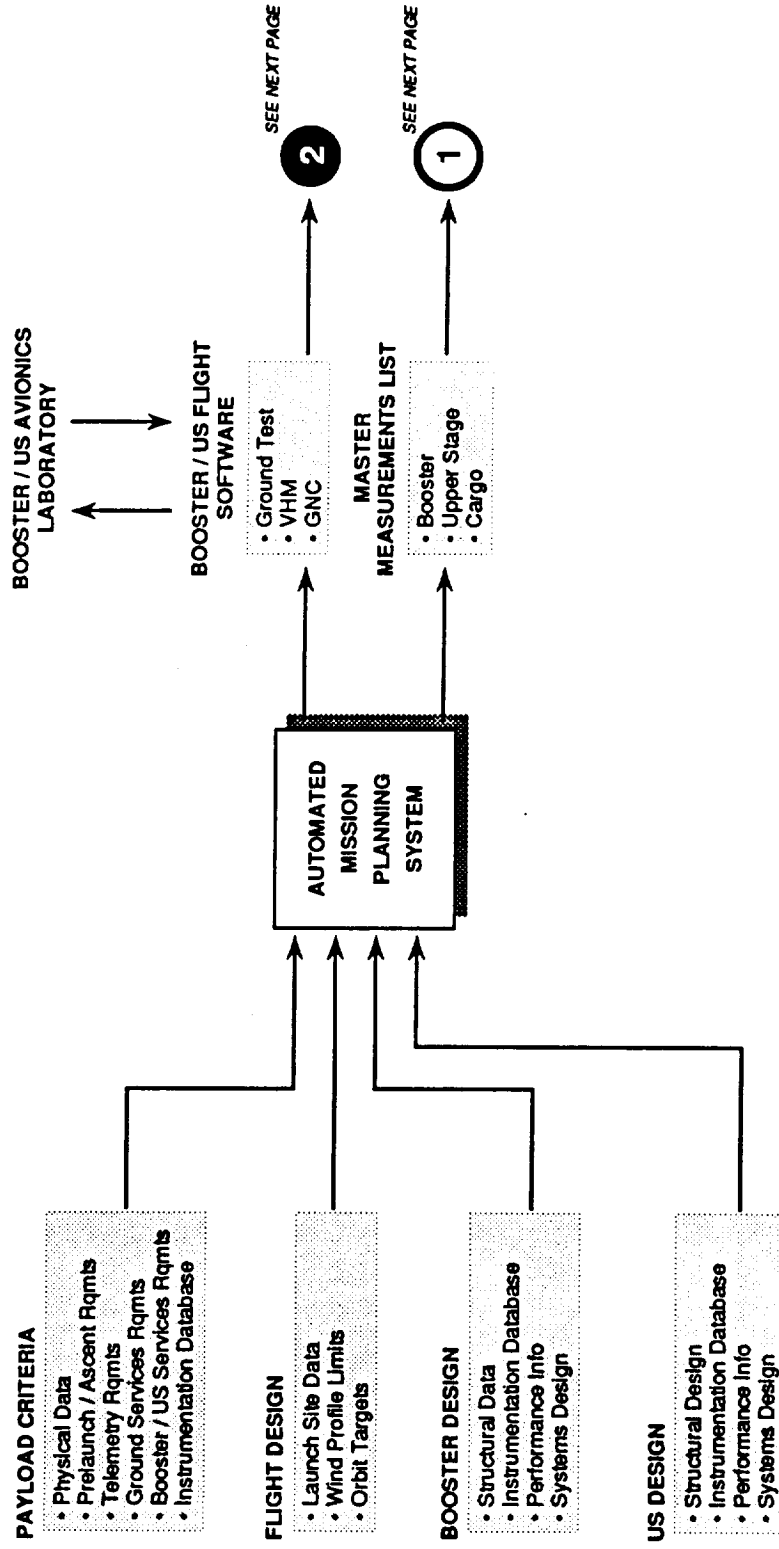
- FMEA / CIL Analysis (Subsystem by Subsystem)
- FDIR Capabilities (Subcomponent Through System)

—Post-Mission

- Data Review Plan (Autonomous / Data Compression)
- Data Archive Plan (Autonomous / Data Compression)

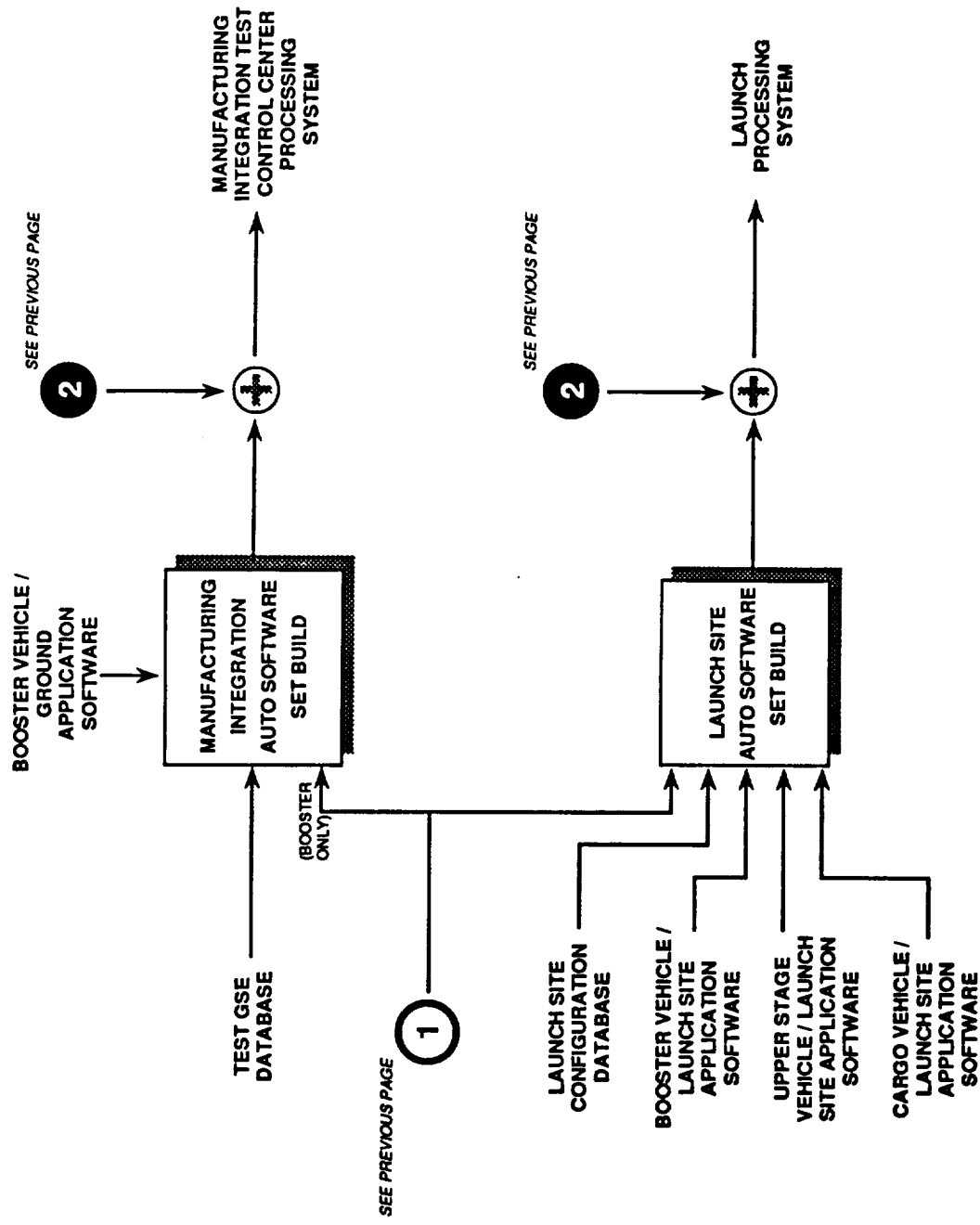
MISSION INTEGRATION ARCHITECTURE

24



MISSION INTEGRATION ARCHITECTURE

25

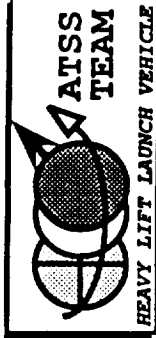




HMS CATEGORIES

VEHICLE

MANAGEMENT



HMS CATEGORIES — VEHICLE MANAGEMENT

27

COMPONENT Define VHM Requirements for Component Life Cycle Phases

- ☐ Engineering Development, Manufacturing and Acceptance Test
- ☐ Component Installation into Vehicle Element and Test
- ☐ Preflight Readiness Verification
- ☐ Launch Commit Criteria
- ☐ Mission Operations

SUBSYSTEM Define VHM Requirements for Subsystem Life Cycle Phases

- ☐ Vehicle Element Manufacturing Test
- ☐ Preflight Readiness Verification
- ☐ Launch Commit Criteria
- ☐ Mission Operations

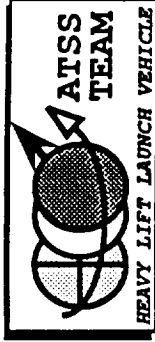
SYSTEM Define VHM Requirements for Subsystem Life Cycle Phases

- ☐ Vehicle Element Manufacturing Test
- ☐ Integrated Vehicle Flight Readiness
- ☐ Launch Commit Criteria
- ☐ Mission Operations

HMS CATEGORIES — VEHICLE MANAGEMENT

COMPONENT VHM Requirements Definition for Component Life Cycle Phases

- ❑ Engineering Development, Manufacturing and Acceptance Test
 - Component Level Internal Redundancy
 - BIT to Subcomponent Level
 - Fault Isolation
 - Factory Development / Evolution of Support Equipment
 - Component Performance Analysis
 - Data Gathering
 - Instrumentation and BIT Validation
 - LRU Maintenance and Repair Procedures
 - Component Performance Design Verification
- ❑ Vehicle Element Component Installation and Test
 - Component Level Checkout Based on BIT
 - Faults Traced Historically Through CALS
 - Depot / Bench Test Procedures / GSE Based on ATP



HMS CATEGORIES — VEHICLE MANAGEMENT

29

COMPONENT

VHM Requirements Definition for Component Life Cycle Phases (cont'd)

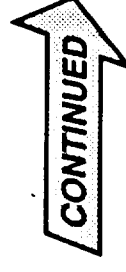
- ❑ Launch Site Preflight Verification and Launch Commit Criteria (LCC)
 - Component Level Health Based on BIT
 - Component Level LCC
 - Minimum Subset of BIT Status Data
 - Functional Criticality
 - Minimum Performance Limits to Support Mission / Margins
- ❑ Mission Operations
 - Component Level FDIR via Internal Hardware / Software Redundancy
 - Component Level In-flight Health Determined BIT and Performance Data
 - Component Health and Performance Critical Information Downlinked
 - Post-Flight Autonomous Data Analysis
 - Post-Flight Data Reduced, Compressed and Archived into CALS

HMS DESCRIPTION FOR VEHICLE LIFE CYCLE PHASES

30

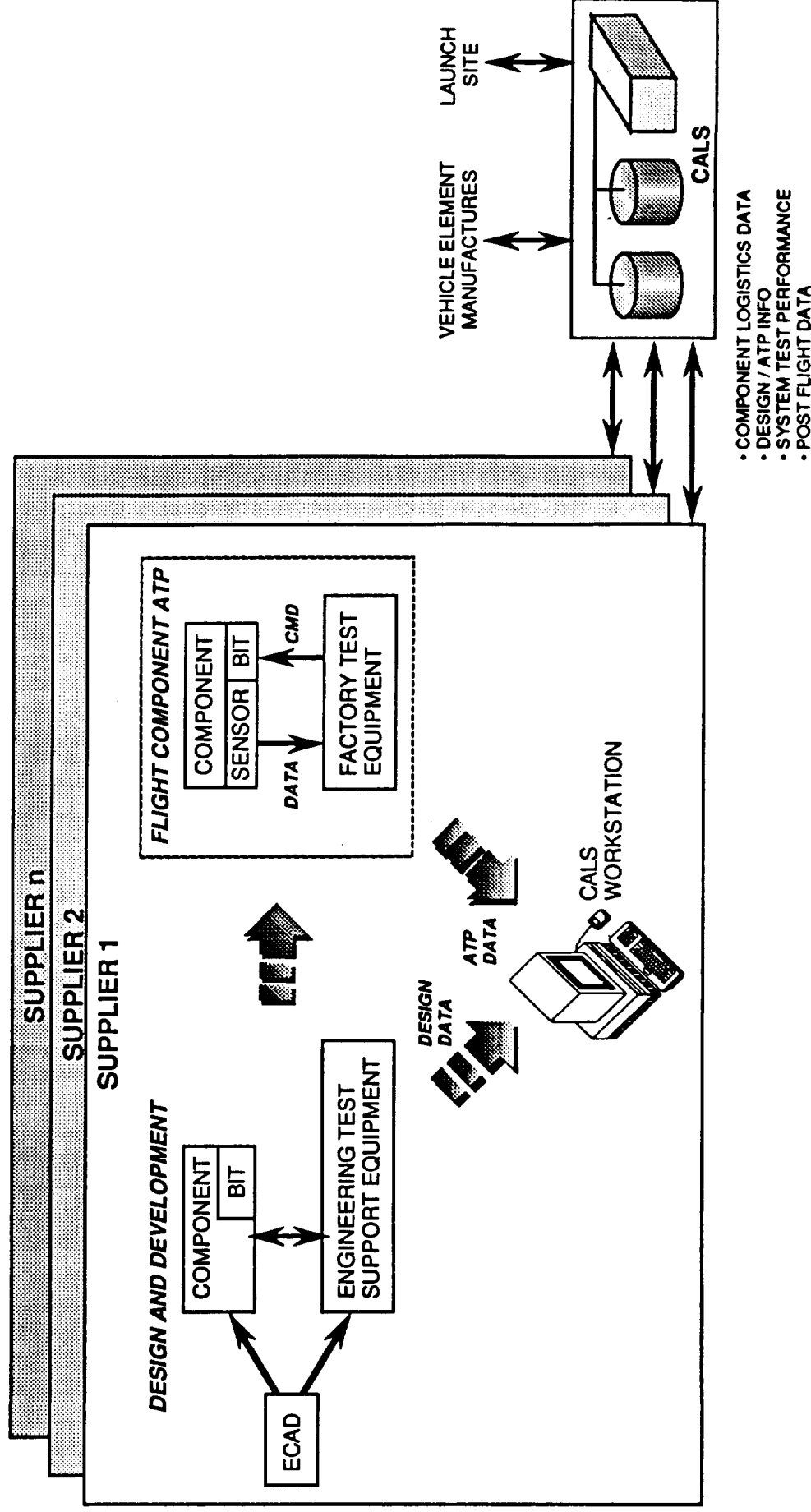
COMPONENT LEVEL: ENGINEERING DEVELOPMENT, MANUFACTURING AND ACCEPTANCE

- ❑ Design Utilizing TQM System Engineering Process and Engineering Computer-Aided Design
- ❑ Support Equipment for Test Evolve During Manufacturing Process and Duplicated for Use at Depot Level Repair / Checkout Site (Vehicle Manufacturer and Launch Site)
- ❑ Acceptance Test Takes Full Advantage of Embedded VHM
- ❑ ATP Data is Compressed, Tagged and Input to CALS for Use During Vehicle Manufacturing and Launch Site Operations
 - Component Performance Data
 - BIT
 - Instrumentation
 - Self Test
 - Component Fault Data
 - Component Logistics Data
 - Fault Isolation Procedures
- ❑ ATP Requirements Are Driven to Program Master Verification Plan



HMS DESCRIPTION FOR VEHICLE LIFE CYCLE PHASES

COMPONENT LEVEL: ENGINEERING DEVELOPMENT, MANUFACTURING AND ACCEPTANCE



HMS DESCRIPTION FOR VEHICLE LIFE CYCLE PHASES

32

VEHICLE ELEMENT SYSTEMS INTEGRATION AND TEST

- ❑ Component Installation and Subsystem Integration
- ❑ Subsystem Component Test Compliance Based on BIT and Component Performance Data
- ❑ Subsystem Level Factory Acceptance Tests
 - Factory Test Requirements Are Driven by Program Master Verification Plan
 - Avionics Subsystems Test Based on Component BIT and Subsystem Performance via Instrumentation
 - Fluid Subsystems Tests Based on Component BIT, Manual Joint by Joint Leak Checks and Automated Pressure Decay Checks
 - Ground Based Software Performs Automated Test Sequences via Operator Menu Select
 - Flight (On-board) Software is Validated / Verified in a Systems Laboratory Against Flight Type Avionics and High Fidelity Propulsion Systems Emulations
 - System Level BIT Built Into Flight Software

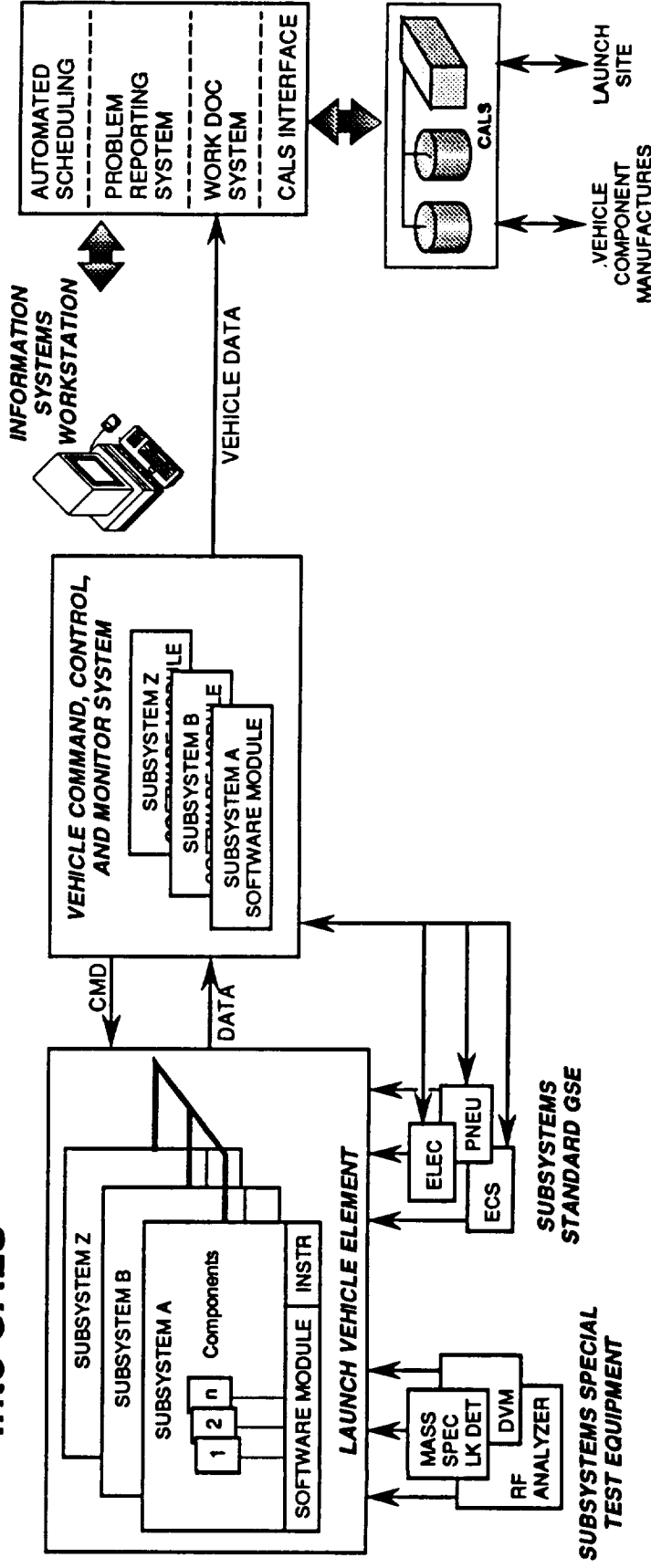
CONTINUED

HMS DESCRIPTION FOR VEHICLE LIFE CYCLE PHASES

33

VEHICLE ELEMENT SYSTEMS INTEGRATION AND TEST (cont'd)

- ❑ Ground Based FDIR
- ❑ Vehicle Subsystems Test Data Are Compressed, Tagged and Input to CALS
- ❑ Vehicle Design, Manufacturing Drawings and Test Procedures Are Maintained in CALS
- ❑ Component Fault Histories Are Automatically Sorted and Filed into CALS





HMS DESCRIPTION FOR VEHICLE LIFE CYCLE PHASES

34

PREFLIGHT VERIFICATION AND LAUNCH COMMIT

- ❑ Launch Site Vehicle Integration, Preflight Readiness and Launch Commit
- ❑ Systems Level Verification
 - Subsystem Component Functional Verification Based on BIT and Performance in System Level Checkout
 - Element to Element to Ground Systems Interface Verification Tests
 - Subsystem Verification Performed by Automated Sequences Executed from CCMS and Vehicle Flight Software
- ❑ Vehicle Systems FDIR Responsibility Shared by Ground Software (CCMS) and Vehicle Flight Software
 - Vehicle Systems FDIR Approaches Total Flight Software Responsibility as Systems Become Flight Ready
 - Ground Software Maintains Ultimate Launch Commit Responsibility
 - Ground Software Maintains Ultimate Command Authority Over Airborne Software Until Launch Commit

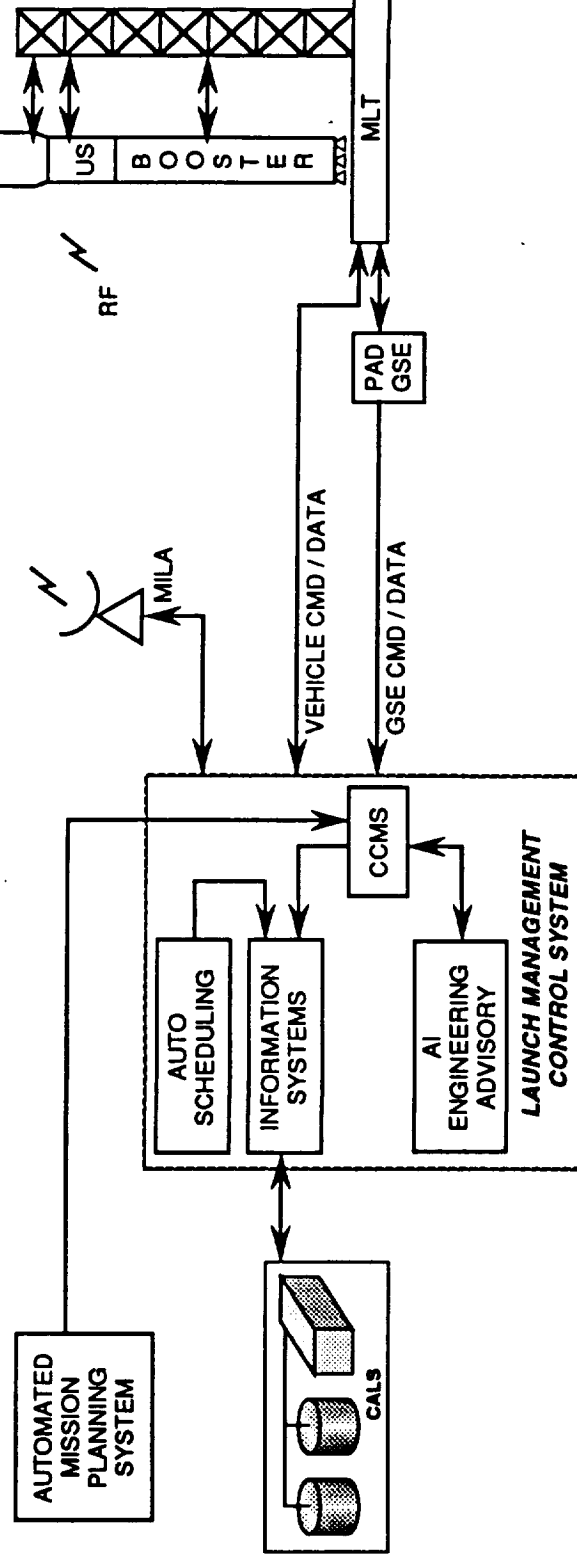


HMS DESCRIPTION FOR VEHICLE LIFE CYCLE PHASES

35

PREFLIGHT VERIFICATION AND LAUNCH COMMIT

- ❑ Autonomous Launch Countdown
 - Ground Software (GLS) Manages All Integrated Ground / Airborne Systems
 - Ground Software Transfers Vehicle Element Subsystem Management to Airborne Software When Flight Ready and Independent from Ground Systems
 - Manual Operations Executed via Software Prompts to the Launch Test Team



HMS DESCRIPTION FOR VEHICLE LIFE CYCLE PHASES

36

MISSION OPERATIONS

- ❑ Vehicle FDIR is Full Internal
- ❑ Downlink Instrumentation Recorded into LMCS
 - Automated Post Flight Data Analysis Utilizing AI and Previous Mission Data
 - All FDIR Events Recorded
 - Faults Categorized by Degree of Significance
 - Reconfiguration Actions
- ❑ Component, Subsystem and Vehicle Element Flight History Data Compressed and Archived into CALS

HMS CATEGORIES

GROUND MANAGEMENT

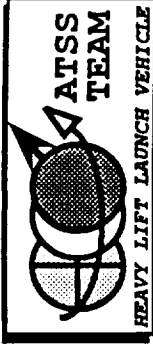
HMS CATEGORIES — GROUND MANAGEMENT

MANUFACTURING TEST GSE

- ❑ Component Factory Test Equipment
- ❑ Commonality Between Suppliers
- ❑ Common Data Format to Information Systems Database (CALS)

VEHICLE ELEMENT MANUFACTURING GSE

- ❑ Factory Integrated Control / Monitor Test System
 - Component, Subsystem, System Auto Test Verification
 - Launch Site / Government Acceptance Verification
 - Provides Integrated Element Data to Information System Database (CALS)
- ❑ Standalone Subsystem Manufacturing Test Equipment
 - Manually Controlled Pneumatic Panels
 - Mass Spectrometer Leak Detectors
 - Manually Controlled Electronic Instruments



HMS CATEGORIES — GROUND MANAGEMENT

LAUNCH SITE GROUND SUPPORT SYSTEMS

- ❑ Launch Processing System (LPS)
 - Central Data System (CDS)
 - Command Control and Monitor System (CCMS)
 - Record and Playback System (RPS)
 - Shared Peripheral Area (SPA)
 - Emergency Safing System
- ❑ Engineering Advisory System
 - Artificial Intelligence Applications Software for Vehicle / Ground Systems Anomaly Notification and Analysis
 - Post-Flight Data Analysis / Compression
 - Real Time Data Analysis
- ❑ Booster Processing Facility
 - Depot for Selected Component Repair / Recertification
 - Factory Test Equipment
 - Core Vehicle Power Up Systems Support
 - Ground Power Supplies
 - ECS Purge
 - Data and Command Interface Links



40

HMS CATEGORIES

INFORMATION

SYSTEMS

HMS CATEGORIES — INFORMATION SYSTEMS

ENGINEERING ADVISORY SYSTEM

- ❑ Vehicle and Ground Systems Data Information Utilizing Artificial Intelligence for Fault Isolation and Corrective Action Recommendations
- ❑ Rapid Access to ARchival and On-line Vehicle Data
- ❑ Operational Test Requirements and Launch Commit Criteria Anomaly Management
- ❑ Launch Operations Management System

OPERATIONS INFORMATION SYSTEM

- ❑ Electrical Connector Analysis and Integrity Network
- ❑ Vehicle / GSE Operations and Maintenance Procedure Library
- ❑ Vehicle Test Requirements and Launch Commit Criteria Documents
- ❑ Real Time Vehicle Schedules
- ❑ Real Time Test Constraints Tracking System

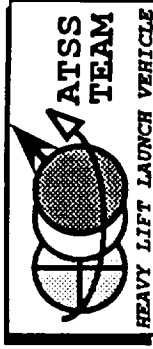
HMS CATEGORIES — INFORMATION SYSTEMS

COMPUTER-AIDED LOGISTICS SYSTEM (CALS)

- ☐ Vehicle / GSE Parts Information (Location, History, Test History, etc.)
- ☐ Problem Reporting and Configuration Accountability (PRACA) System
- ☐ Vehicle / GSE CAD Drawing Files

INTEGRATED PROCESSING CONTROL SYSTEM

- ☐ Automated Requirements Management System
- ☐ Work Preparation Support System
- ☐ Shop Floor Control / Data Collection System
- ☐ Computer-Aided Scheduling and Planning of Resources



INTEGRATED HMS FOR ATSS

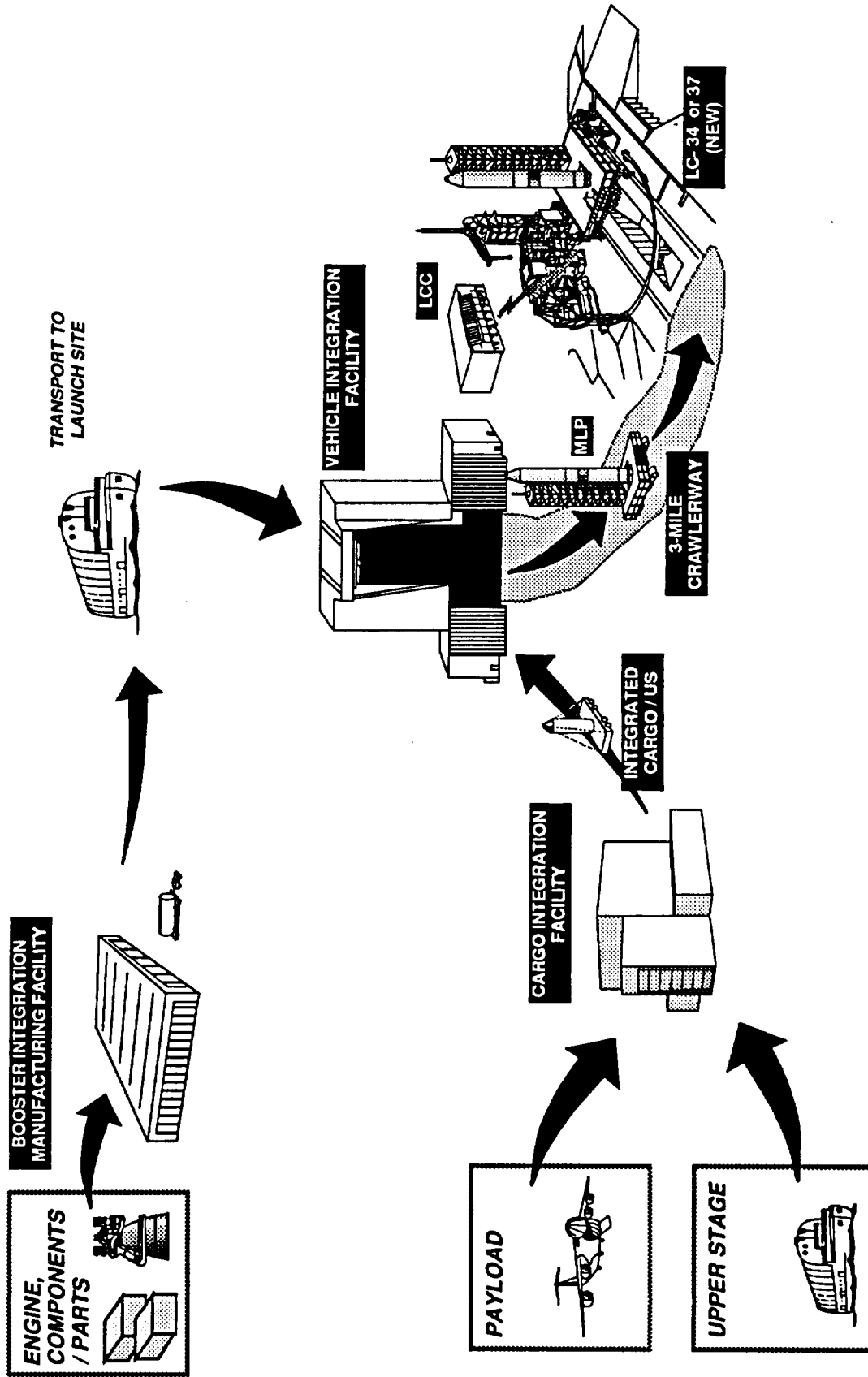
The following is an attempt to describe HMS / VHM for a medium lift (50Klb to LEO) booster using the previously described concepts. A generic liquid propellant (LO₂ / LH₂) booster design is analyzed from component delivery and manufacture through launch site integration and post-mission analysis. High energy upper stage and payload integration are included in the study for booster and launch site interfaces.

Engineering analysis of fault tolerance, time to failure, BIT technology, and other detailed design issues are not presented. Rather, an approach of operations analysis is performed, since operations robustness and program efficiency are the ultimate goals of improved HMS technology.



ATSS Vehicle Health Management System Definition Study

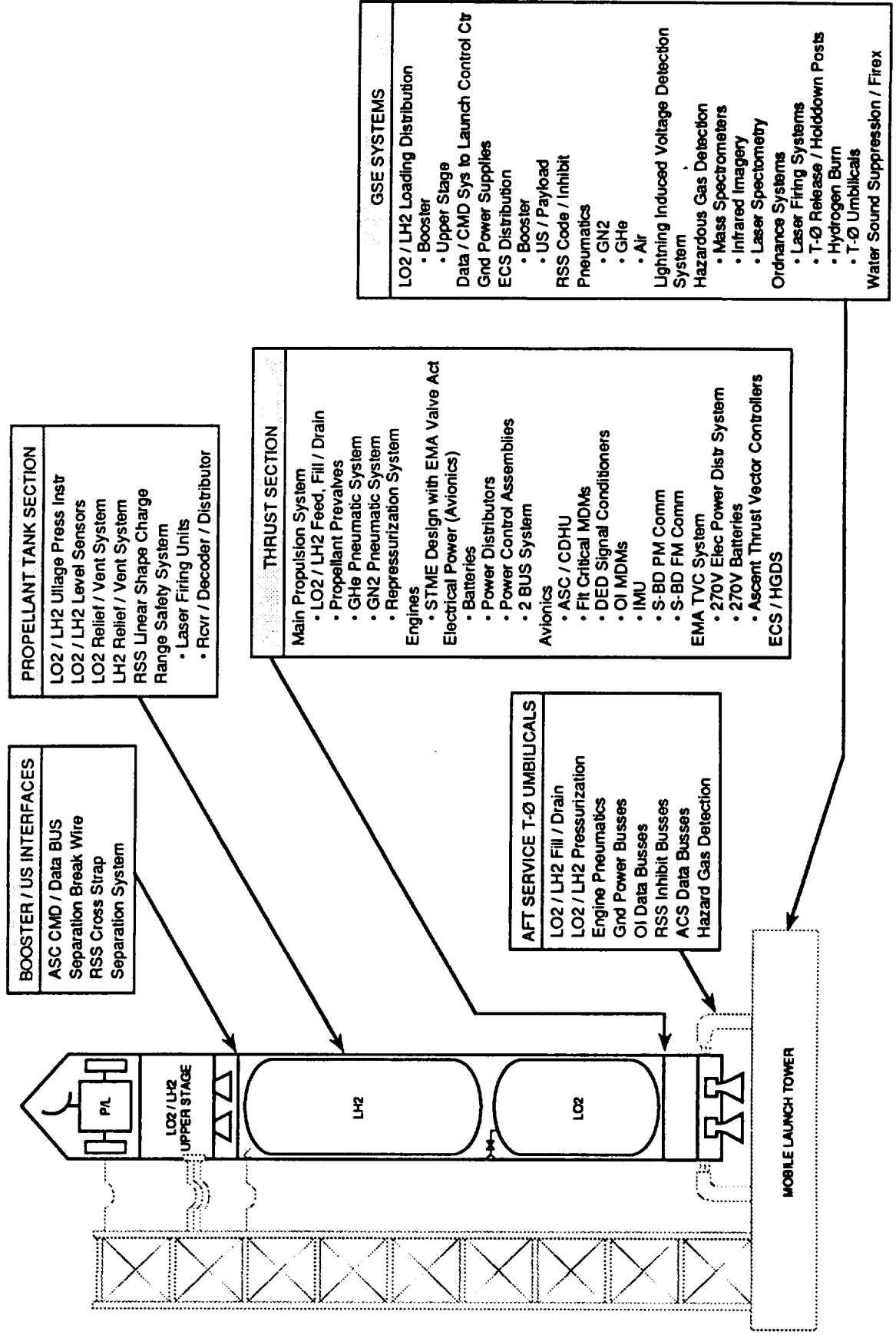
ATSS MEDIUM LIFT GENERIC LAUNCH VEHICLE PROCESSING SCENARIO

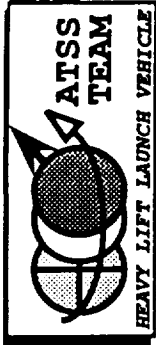




ATSS Vehicle Health Management System Definition Study

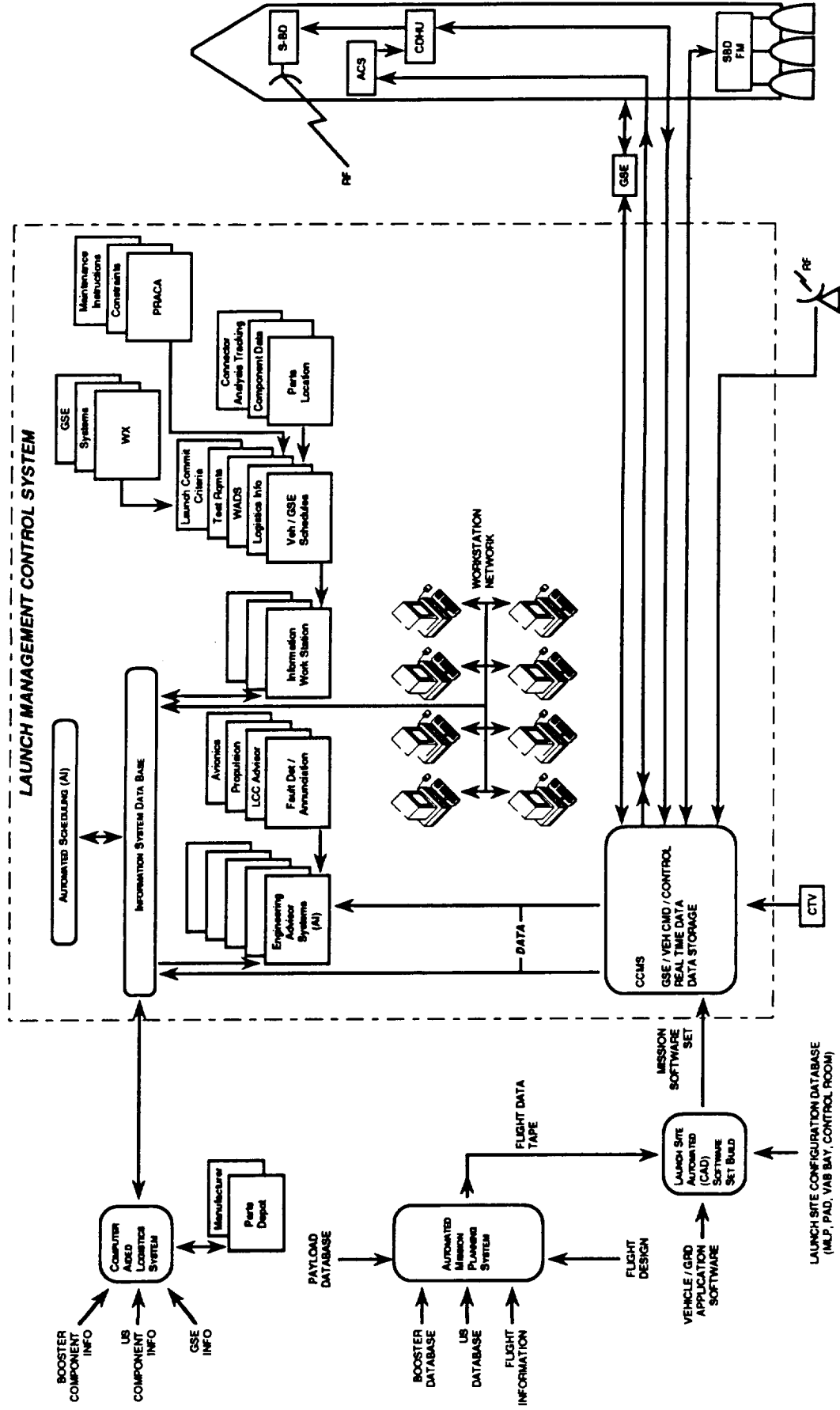
ATSS MEDIUM LIFT GENERIC LAUNCH VEHICLE DESIGN

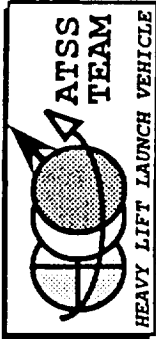




ATSS Vehicle Health Management System Definition Study

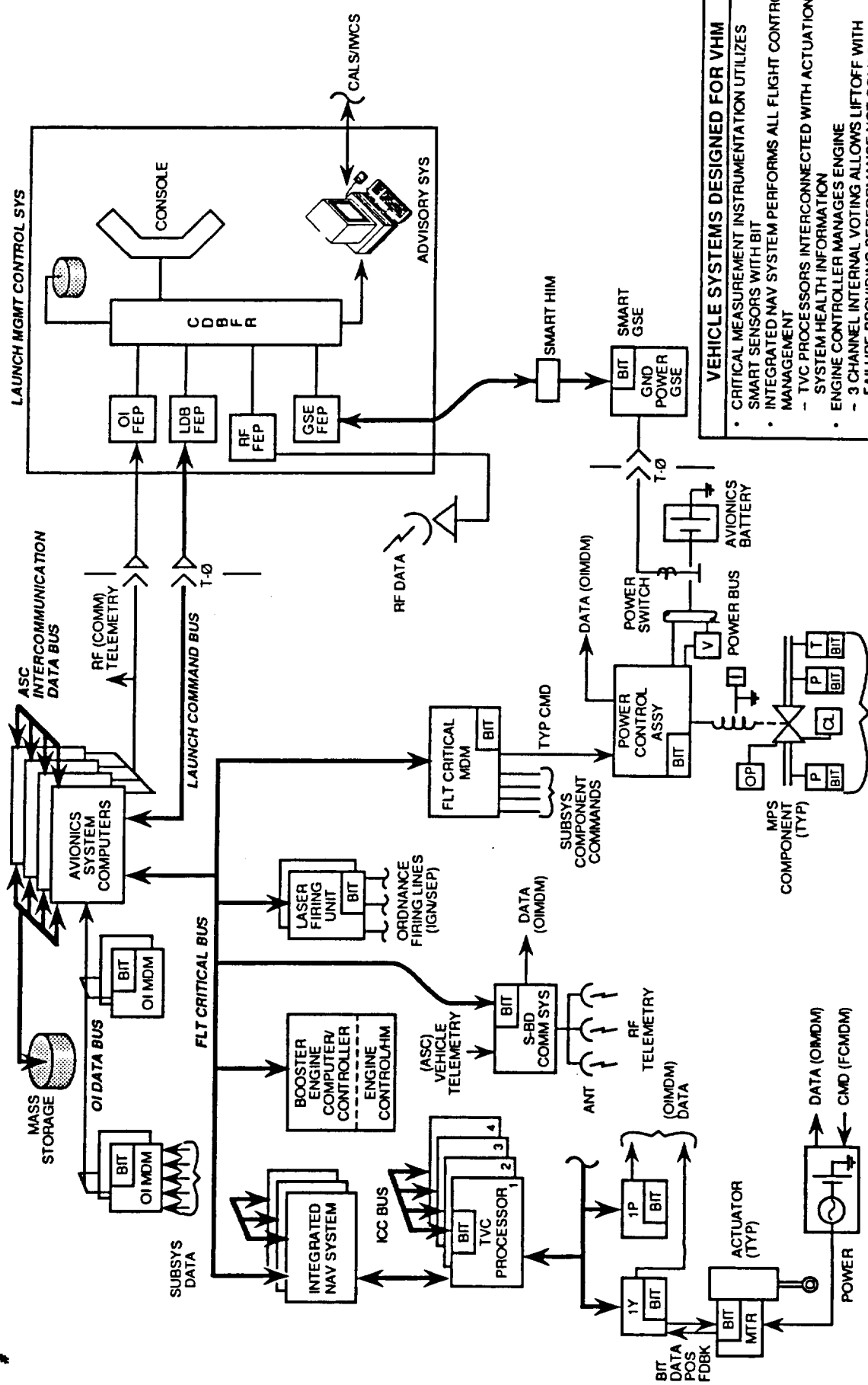
FULLY INTEGRATED LAUNCH SITE HEALTH MANAGEMENT SYSTEM ARCHITECTURE





ATSS Vehicle Health Management System Definition Study

VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

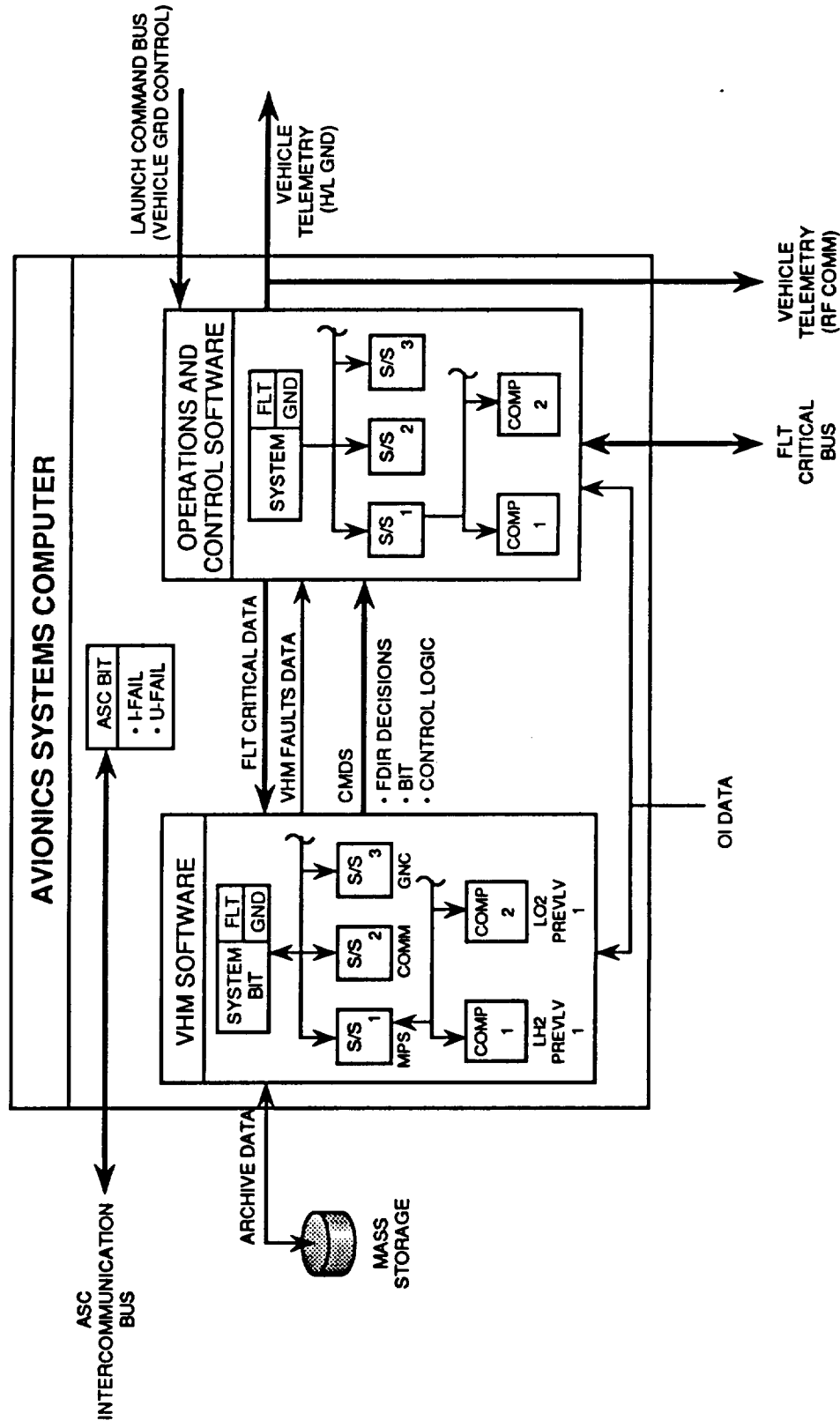


- ### VEHICLE SYSTEMS DESIGNED FOR VHM
- CRITICAL MEASUREMENT INSTRUMENTATION UTILIZES SMART SENSORS WITH BIT
 - INTEGRATED NAV SYSTEM PERFORMS ALL FLIGHT CONTROL MANAGEMENT
 - TVC PROCESSORS INTERCONNECTED WITH ACTUATION
 - SYSTEM HEALTH INFORMATION
 - ENGINE CONTROLLER MANAGES ENGINE
 - 3 CHANNEL INTERNAL VOTING ALLOWS LIFTOFF WITH FAILURE PROVIDING PERFORMANCE NOT COMPROMISED
 - ASC PERFORMS SYSTEM MANAGEMENT
 - INTERFACE TO GND CONTROL (LMCS)
 - SUBSYSTEMS OPERATIONS AND CONTROL
 - VHM BASED ON SENSOR, SUBSYSTEM PERFORMANCE, BIT, TREND DATA AND SOFTWARE



VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

AVIONIC SYSTEM COMPUTER (ASC) SOFTWARE ARCHITECTURE



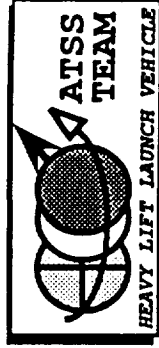


ATSS Vehicle Health Management System Definition Study

VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

TOP DOWN SYSTEMS DESIGN — DESIGN TO OPERATIONS

- ❑ **Common GSE for Component Vendors and Depot Level Maintenance at Launch Site**
- ❑ **Modular Component Avionics Built Into Easy Access Chassis**
 - **Minimizes Electrical Connectors**
 - **Avionics Components Verification Test Entirely via Built-In Test**
 - **Minimizes Component Failure Impact, Replacement and Retest**
- ❑ **Fiber Optic Data Transfer**
 - **Supports Large Band Width of Data**
- ❑ **Minimize System Production Breaks and Vehicle Element Interfaces**
 - **Interface Verification Test Between Vehicle Elements via BIT**
 - **Modular Subsystems Interconnected with Data Buses**



ATSS Vehicle Health Management System Definition Study

VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

MODULAR ON-BOARD VHM SOFTWARE WITH SYSTEM, SUBSYSTEM AND COMPONENT HIERARCHY

- ❑ System Level Ground Test and Flight Health Management
 - Reactive Control Logic Safing Procedures
 - Prerequisite Control Logic Safing Procedures
 - Menu Select for Ground Checkout/Prelaunch Activation
 - Avionics Subsystems Activation and Selftest
 - Fluid/Propellants Subsystems Electro/Mechanical Component Verifications
 - Vehicle Propellant System Management Prelaunch Augmented with Ground Software Communications
 - Ground Capability to Override/Bypass Any Vehicle Health Management Functions (Deselect Failed Instrumentation, Components, etc.)



VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

MODULAR ON-BOARD VHM SOFTWARE WITH SYSTEM, SUBSYSTEM AND COMPONENT HIERARCHY (cont'd)

- ❑ Automatic Component/Subsystem/System Health Monitor
 - Logs Any Fault Data to Downlink Data (Telemetry) Bus
 - Verifies Proper Instrumentation Response During Component Operation
 - Microswitch Activation
 - Voltage Response
 - Valve Timing
 - Valve Solenoid Current Signature
 - Component Built In Test Status Word
 - Pressure, Temperature, Flow Rate Response
- ❑ Provides LMCS with Built-In Test Capabilities via Menu Select (All Commands Issued via Operations and Control Software)
 - Sequenced Systems Activation and Tests
 - Sequenced Subsystems Activation and Tests
 - Sequenced Component Activation and Tests



VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

MODULAR ON-BOARD VHM SOFTWARE WITH SYSTEM, SUBSYSTEM AND COMPONENT HIERARCHY (cont'd)

❑ Launch Countdown

- Ground Launch Sequencer (GLS) Activates Vehicle Subsystems/Components, Commands Built-in Test Sequences and Verifies Flight Ready; Vehicle Subsystem Management is Then Transferred Over to the On-board VHM System for Subsequent Monitor/FDIR for Remainder of Launch Countdown/Mission
- On-board Vehicle Launch Sequencer (Operations and Control Software) Initiates All Vehicle Functions Necessary to Orchestrate Systems Arming, Final Flight Readiness, Engine Start Sequencing, and Launch Commit/Launch Release
- VHM Software Monitors Vehicle Systems Launch Commit Criteria and Automatically Issues LCC Contingency Sequences (via Operations and Control Software) to Verify Flight Acceptability During Anomalous Conditions

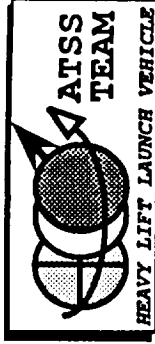


VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

MODULAR ON-BOARD VHM SOFTWARE WITH SYSTEM, SUBSYSTEM AND COMPONENT HIERARCHY (cont'd)

❑ Launch Countdown (cont'd)

- Launch Cutoff/Abort Would be Initiated by VHM Direct or VHM via Loss of "GLS Go for Launch" Flag as a Result of Engine, Vehicle Systems, Payload, Range Safety, or Ground Systems LCC Violation
- Payload LCC Monitored via LMCS Payload Applications Software with Flag to GLS
- GLS Issues Hold/Cutoff Signal for Manual Request, Payload and Ground Systems LCC Violations
- Launch Cutoff/Abort Sequencing/Safing Accomplished as a Shared Response Between GLS and Ops/Control Software with Advisement from VHM Software



ATSS Vehicle Health Management System Definition Study

VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

MODULAR ON-BOARD VHM SOFTWARE WITH SYSTEM, SUBSYSTEM AND COMPONENT HIERARCHY (cont'd)

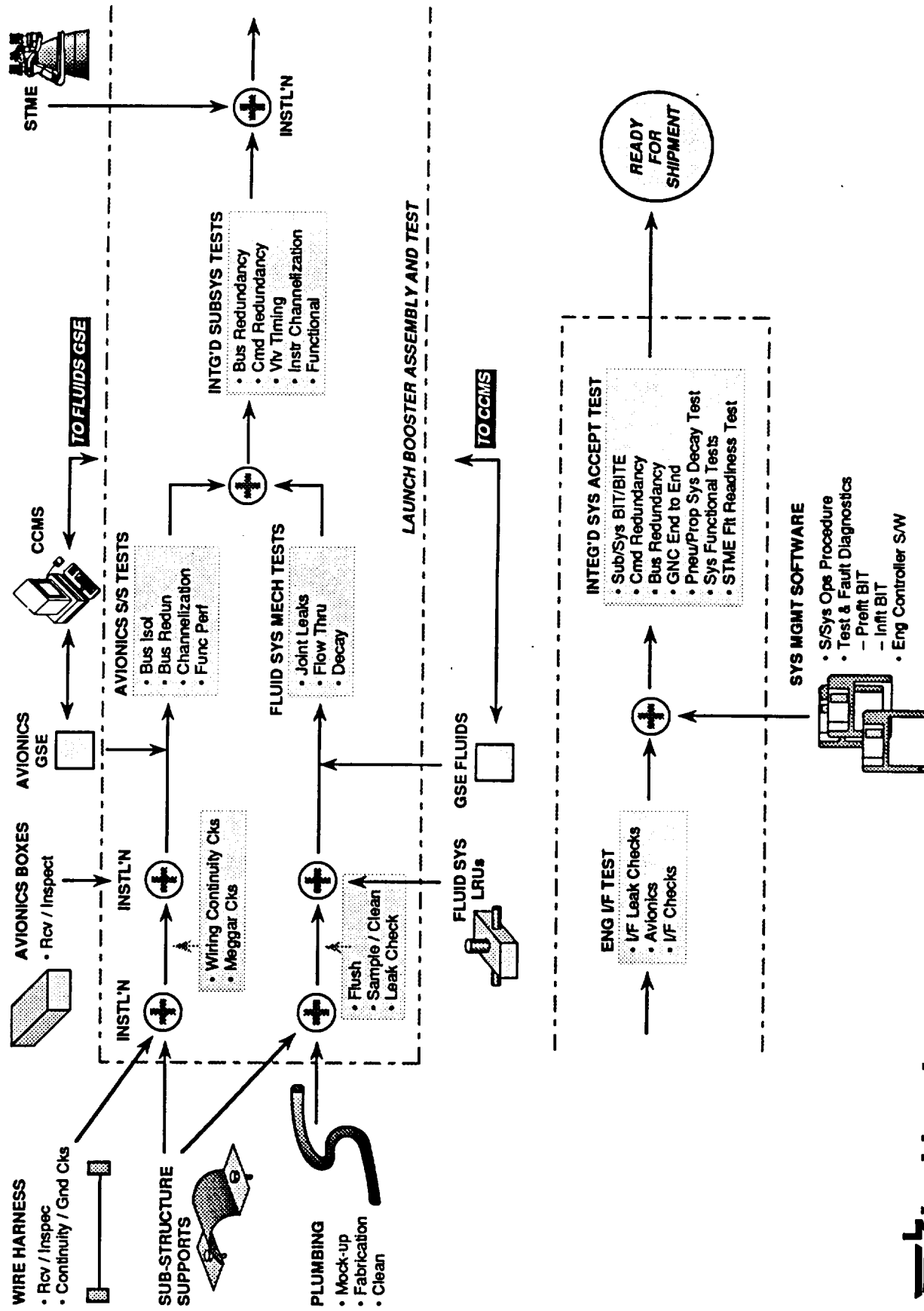
In Flight

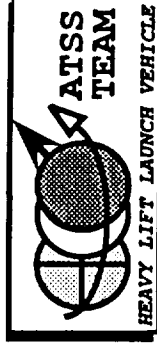
- Vehicle Systems/Subsystems Management via FDIR of Critical Components and Data to Accomplish Mission
- All Vehicle Data is Supplied to the Ground Launch Management Control System for Post Flight Reduction, Compression and Storage
- Utilize On-board Subsystem Built-in Test Software to Aid In-flight Health Management Decisions
- Operations and Control Software Performs Vehicle/Mission Management and Executes VHM-derived Actions; GNC Subsystem is Managed by Software Residing and Executing Within the Integrated Navigation System Processors



ATSS Vehicle Health Management System Definition Study

LAUNCH BOOSTER MANUFACTURING PROCESS

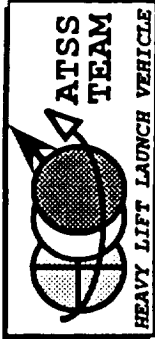




LAUNCH BOOSTER MANUFACTURING PROCESS

SUBSYSTEM ASSEMBLY

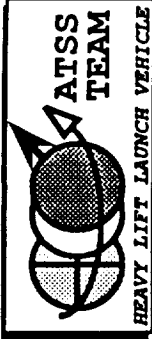
- ❑ Until the Avionics Subsystem is Capable of Supporting Data/Command Capability, Fluids Systems Testing is Limited to Partial Subsystem Overpressure Pneumastat, Joint-by-Joint Leak Checks, and Flow Path Checks. Local GSE is Required for Valve Actuations, Instrumentation and Pressurization
- ❑ Instrumentation Channelization. Smart Sensor Instrumentation May Be Verified via BIT, Although Complete End-to-End Channelization Will Still Require Some Form of Physical Stimulus at the Transducer. All Other Instrumentation is Verified by Physical Stimulus During Subsystems Test
- ❑ Detailed Fluid Subsystems. Requires Operational Instrumentation, Data Management and Electrical Power Systems Active to Support Most Fluid Subsystems Component Verifications. Software to Allow Manual Select Operation of Individual Components and Automated Subsystems Functional Tests are Performed from the Ground Test Processing System. A Separate (Ground-based) Engineering Advisory System Utilizing Artificial Intelligence Software Provides Information to Expedite Fault Detection, Isolation, and Resolution. VHM for Fluids Subsystems are Supported by Limit Switches, Temperature Transducers, Pressure Transducers, MDM Channel BIT, Power Bus Voltage, Valve Solenoid Current and Supporting Avionics Systems General Health Information



LAUNCH BOOSTER MANUFACTURING PROCESS

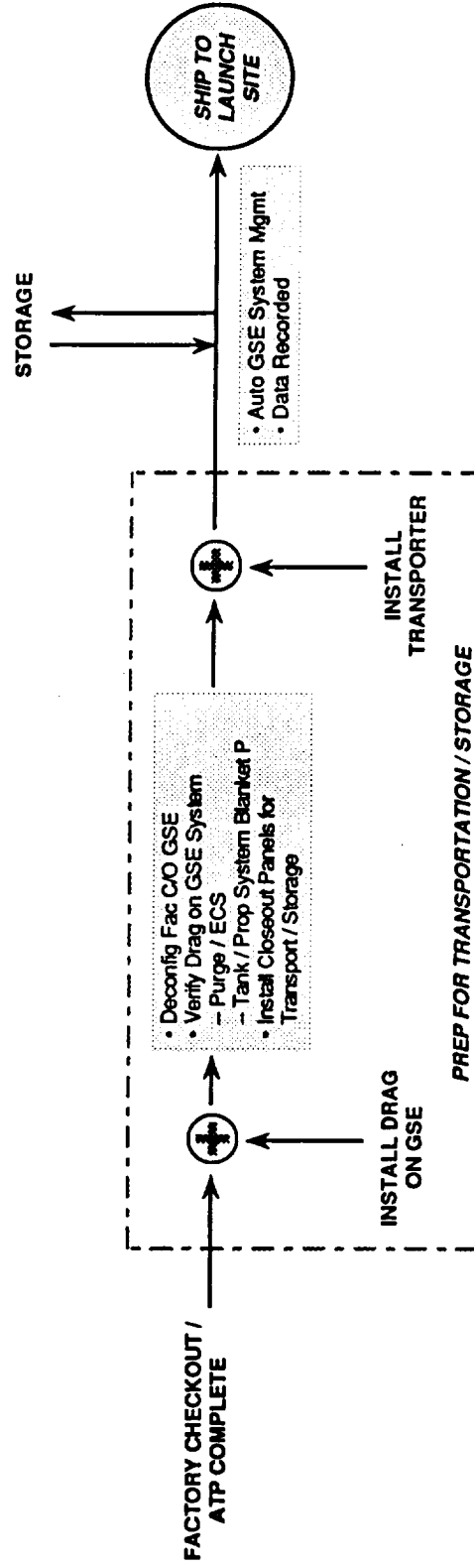
MANUFACTURING INTEGRATION SUPPORT

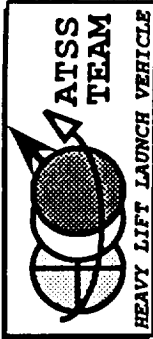
- ❑ All Schedules for Manufacturing, Assembly, Tests and Material Requirements are Generated Utilizing Expert Systems Software
- ❑ Computerized/Robotic Logistics System Delivers Kitted Materials to Shop Personnel as Notified by the Automated Work Control System
- ❑ Vehicle Assembly Drawings and Procedures are Available from CALS for Workstation Viewing, Hardcopy Print Out or Video Recording
- ❑ VCR with Remote RF Transmission Available for Technicians Working in a Hands-Free Environment via Eye Retina Display System
- ❑ Connector Configuration Analysis Network (CCAN) is Utilized to Track the Realtime Vehicle Avionics and Fluids Systems Configuration for Each Launch Vehicle Element; Expert Systems Software Supports as an Aid for Efficient Troubleshooting of Avionics Systems Problems and Vehicle Systems Test Readiness Configurations
- ❑ Bar-Code Scan Utilized for Configuration Management of Parts and Procedures



ATSS Vehicle Health Management System Definition Study

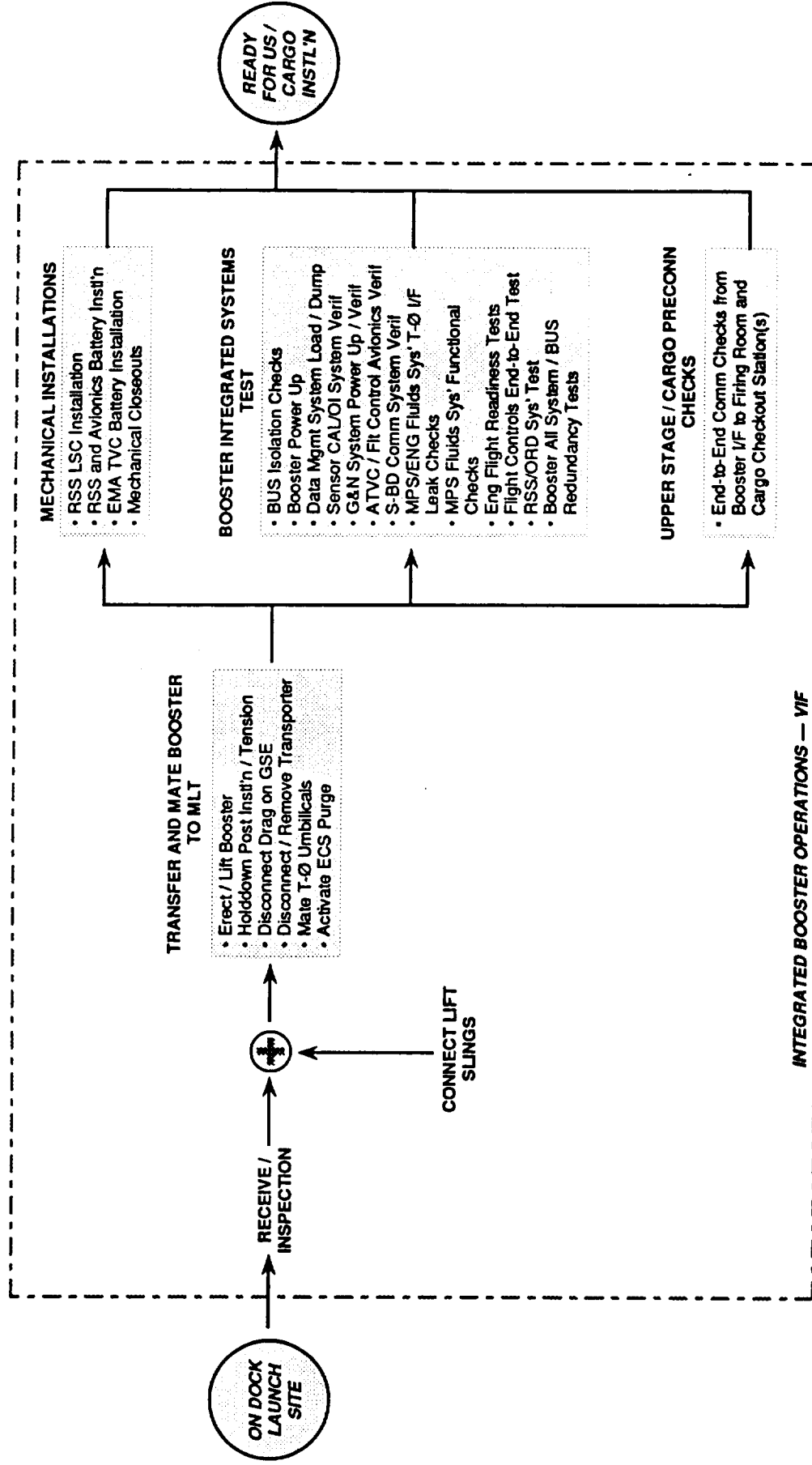
LAUNCH BOOSTER MANUFACTURING PROCESS





ATSS Vehicle Health Management System Definition Study

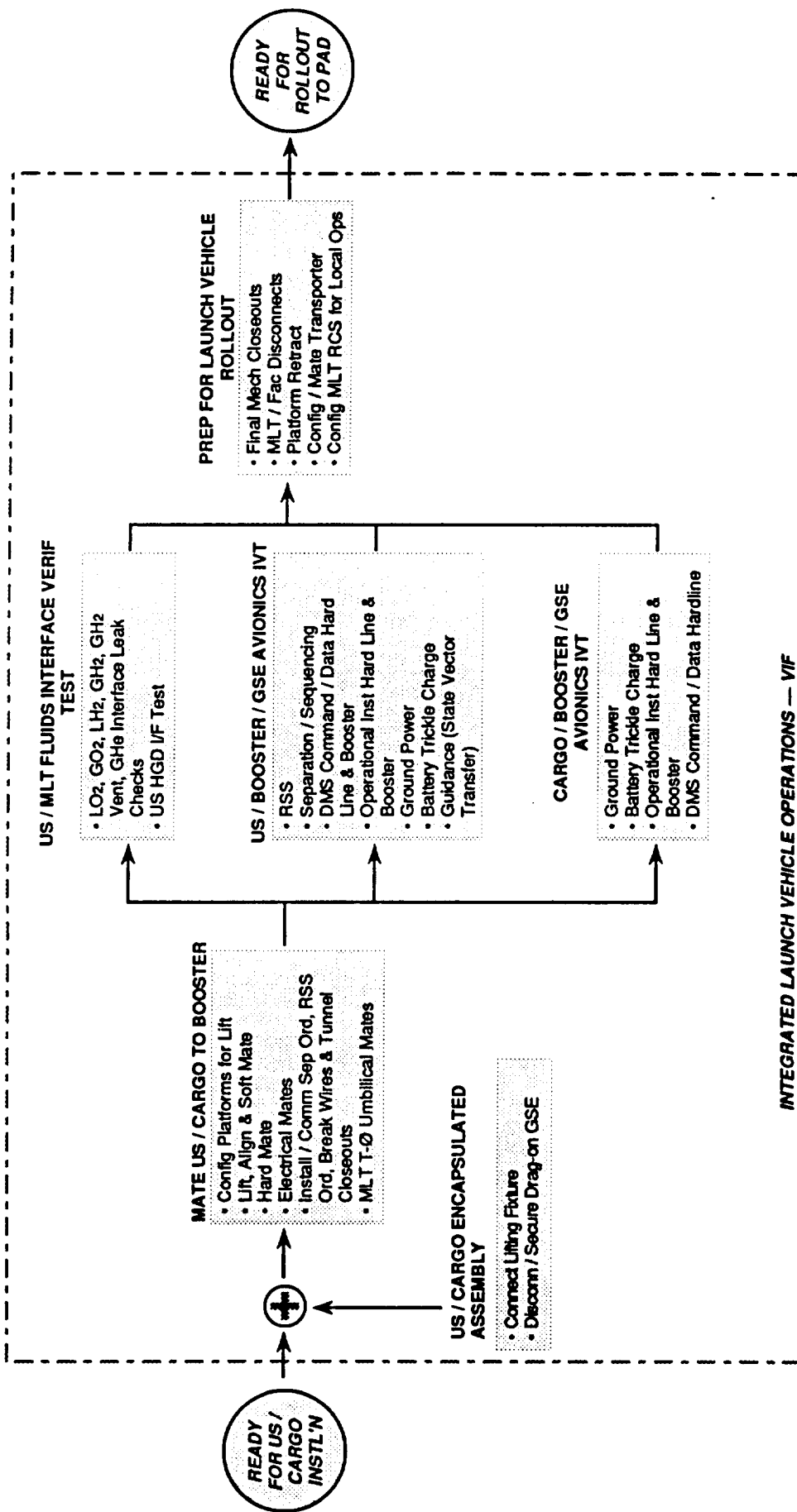
LAUNCH SITE OPERATIONS





ATSS Vehicle Health Management System Definition Study

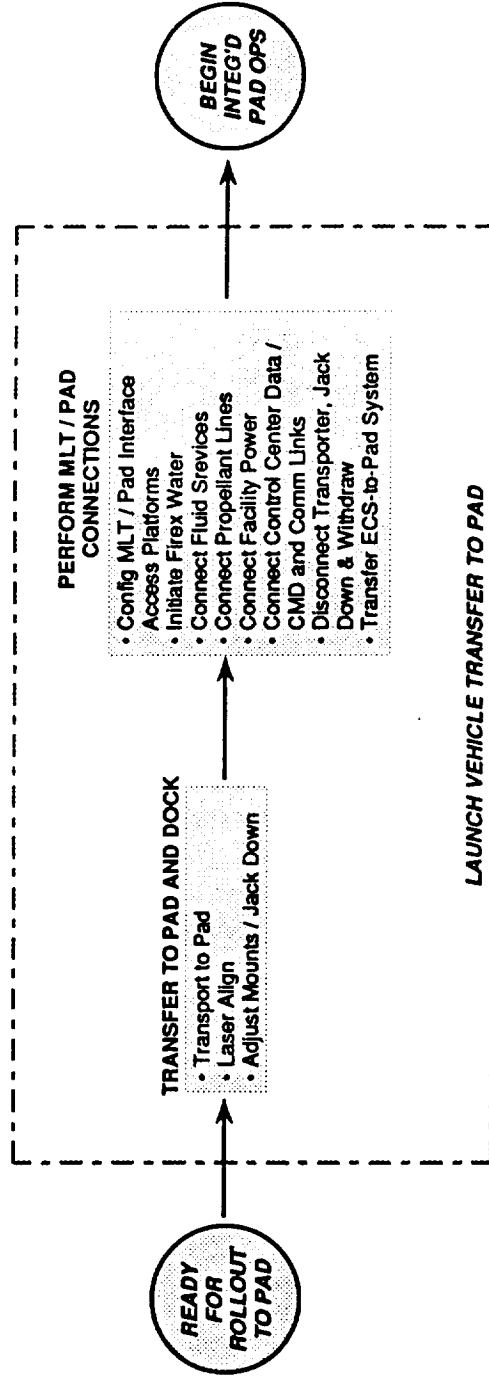
LAUNCH SITE OPERATIONS





ATSS Vehicle Health Management System Definition Study

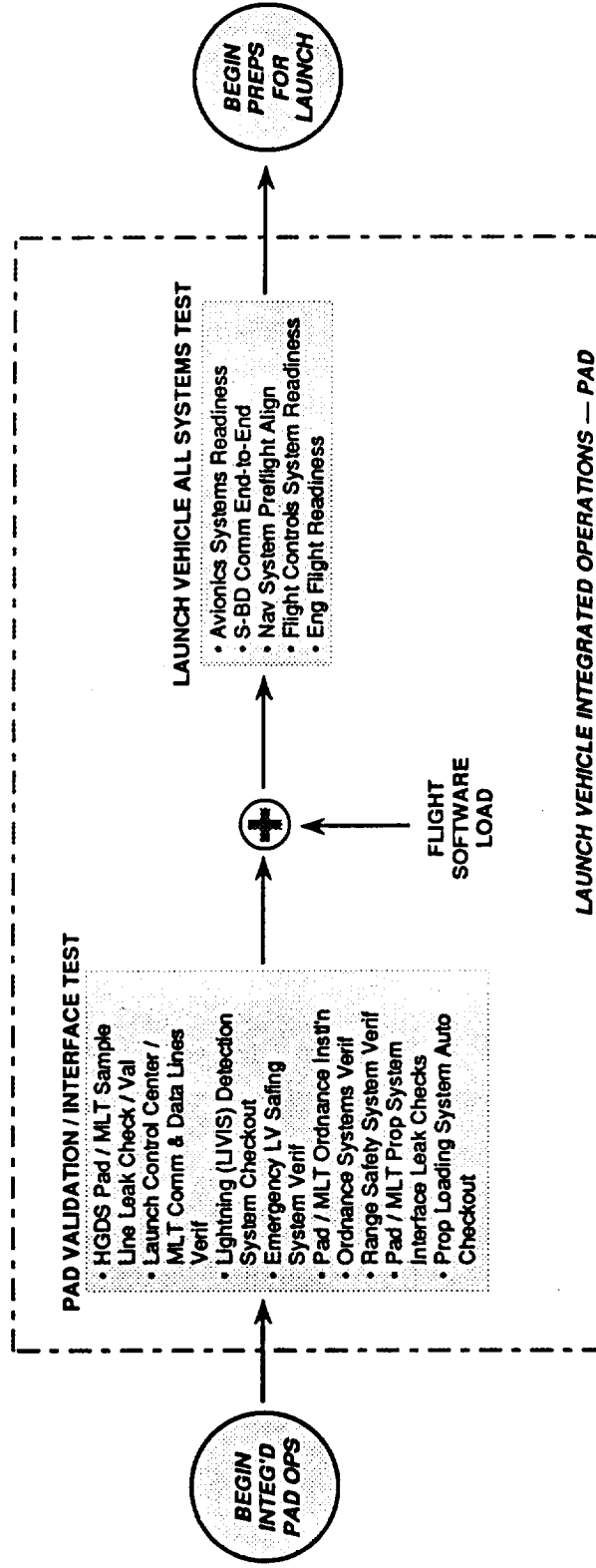
LAUNCH SITE OPERATIONS





ATSS Vehicle Health Management System Definition Study

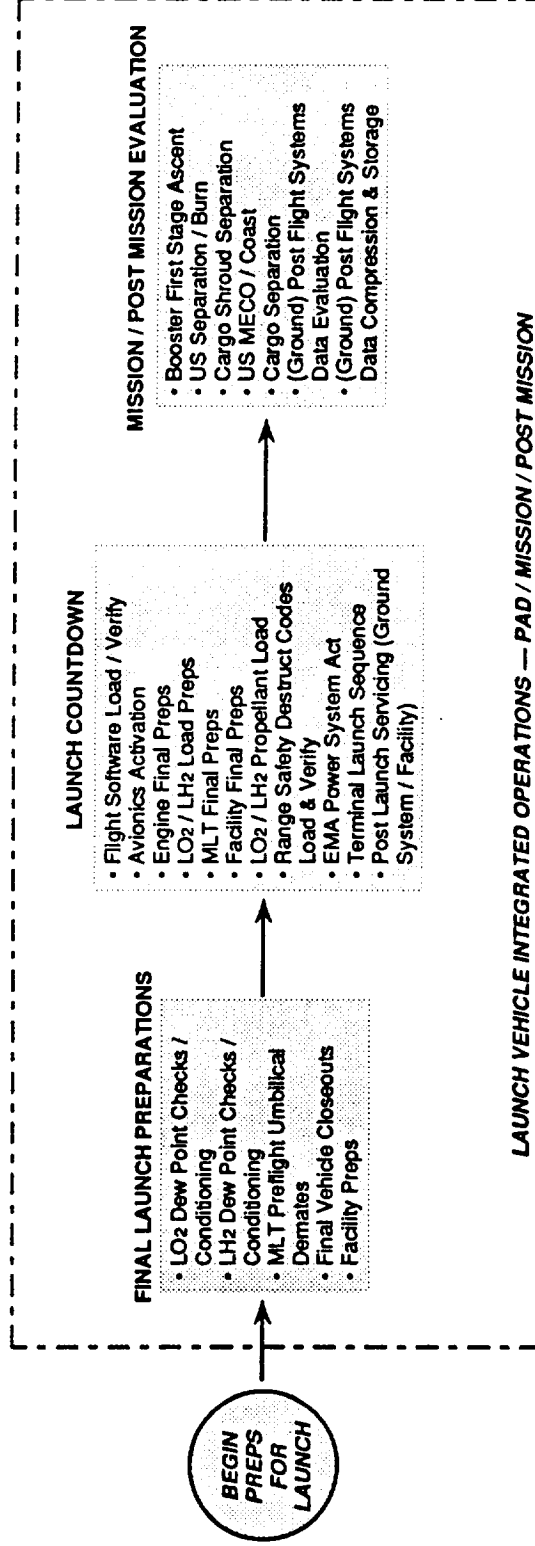
LAUNCH SITE OPERATIONS

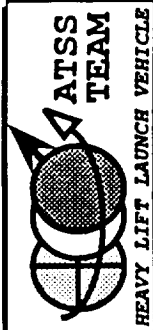




ATSS Vehicle Health Management System Definition Study

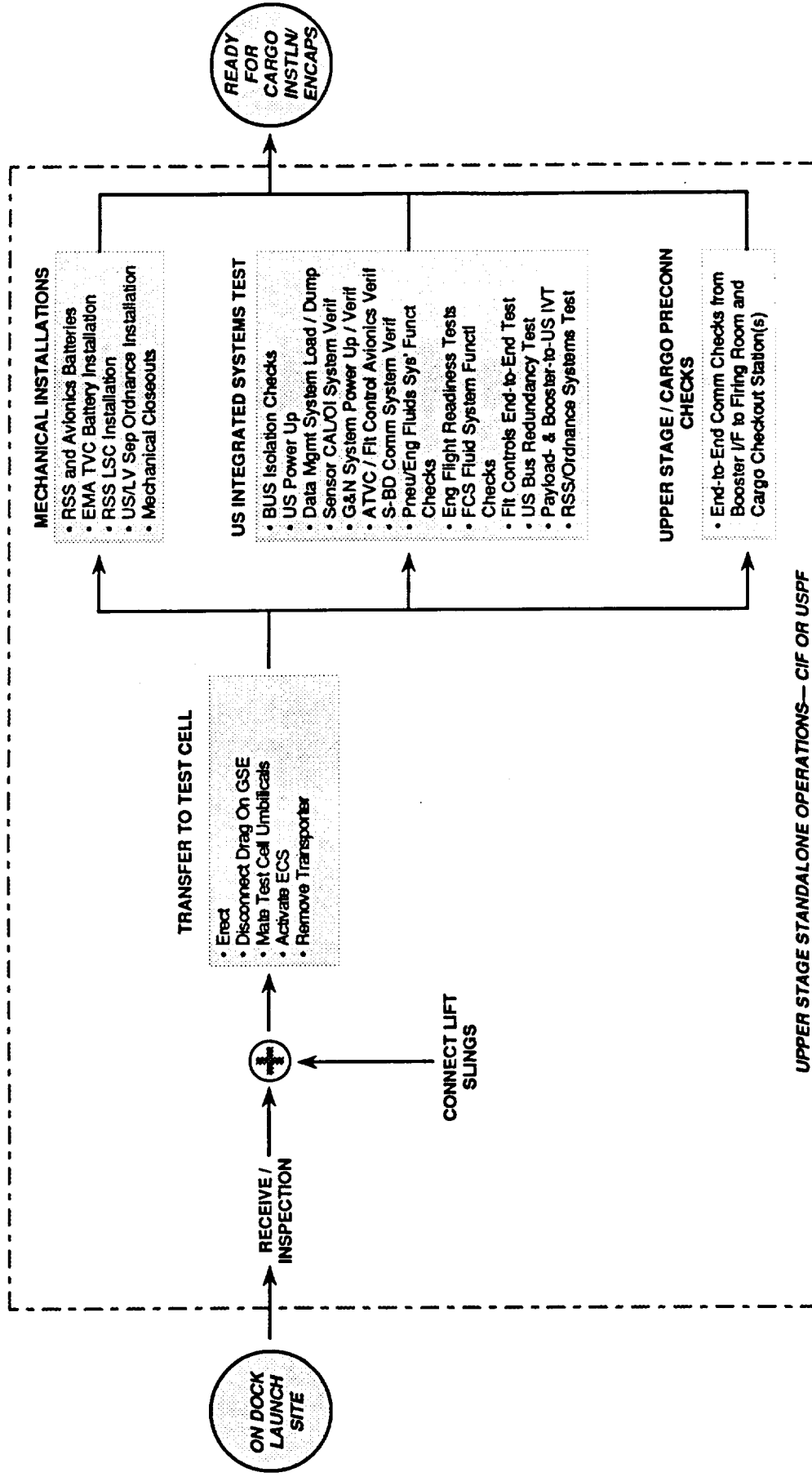
LAUNCH SITE OPERATIONS

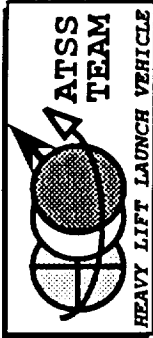




ATSS Vehicle Health Management System Definition Study

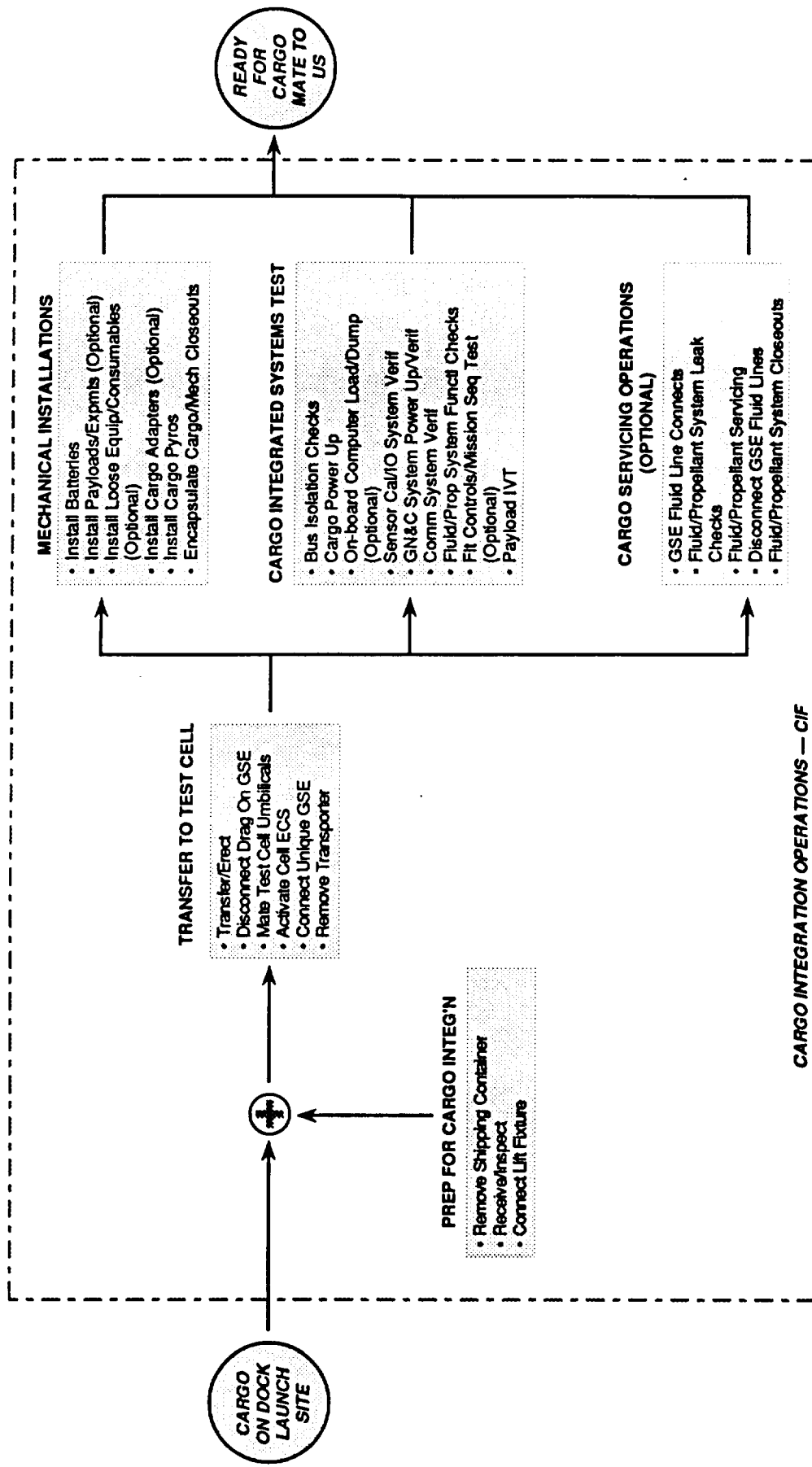
LAUNCH SITE OPERATIONS UPPER STAGE

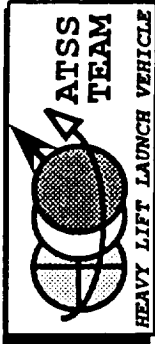




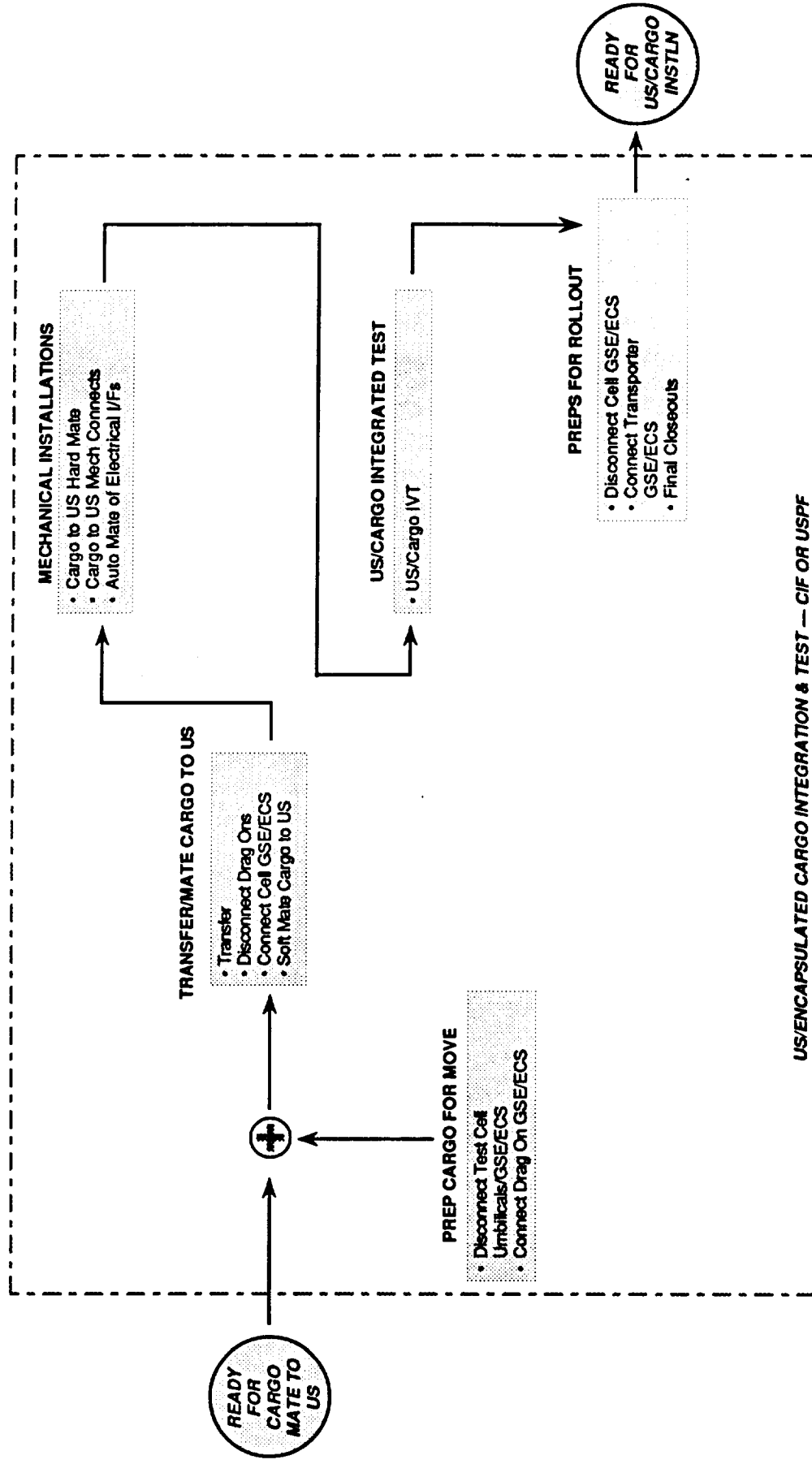
ATSS Vehicle Health Management System Definition Study

LAUNCH SITE OPERATIONS CARGO





LAUNCH SITE OPERATIONS CARGO/UPPER STAGE





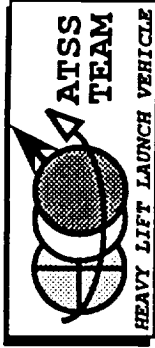
LAUNCH SITE OPERATIONS

PAYLOAD SPECIFIC DESIGN ASSUMPTIONS

- ❑ Standard I/Fs (Standard Cable/Harnesses)
- ❑ Standard Services (Fluids, Power, Optics, Data, etc.)
- ❑ Standard Locations (Fixed Design Location and Breakout Points)

PAYLOAD GENERIC GROUND TEST EQUIPMENT

- ❑ Utilize the Common Test Equipment (CTE) Approach
- ❑ Standard Cable I/Fs
- ❑ CTE Software Driven
- ❑ Software Automatically Created from Mission-Unique Engineering via Engineering Knowledge Base
- ❑ Automatic Test of I/Fs and Functions
- ❑ Automatic Problem Isolation
- ❑ Manual Operator Capabilities
- ❑ Master Gauges/Payload Adapters for Factory Use



LAUNCH SITE OPERATIONS

PAYLOAD OPERATIONS

- ☐ Paperless System Using Hand-Held Computers with Bar-Code Readers to Cut Down Errors in Procedures and Accuracy of Buys/Event Timing; Computer Links RF or Optical
- ☐ Automated Mating of Interfaces Mechanical/Electrical Possible if Standard I/Fs and Locations Maintained
- ☐ Minimum Standard Access Panels for On-MLT Ops
- ☐ Single Payload Umbilical Panel with All Services Required
- ☐ Acquire Payload Health Monitoring Through Payload T-Ø Umbilical, Not Through Launch Vehicle
- ☐ Payload In-Flight Data Should Come Through Payload Comm to Shroud Antenna System (S-Band)
- ☐ Payload Ground Power Should Come Through Payload T-Ø and Be Dedicated (with UPS Capability)
- ☐ Launch Vehicle/US Interfaces to Payload Should Only Be Breakwire/SEP Commands or RSS
- ☐ Shroud/Payload ECS Instrumentation Should Be Complete and Through T-Ø
- ☐ Non-Intrusive Test and Troubleshooting Should Be Built-In So Fault Isolation During Ground Test Does Not Require Unnecessary Connector Demates and Retest

12.0 Major TA-2 Presentations

Figure 12.0-1 summarizes the titles of the major presentations given during the course of the TA-2 contract.

1992 Presentations:
• "Lowering the Operations Cost of New Launch Vehicle Systems", ECON, February 11, 1992
• "First Lunar Outpost Heavy Lift Launch Vehicle Development, NLS-Derived Configuration Assessment", work performed for MSFC by Lockheed during February - March 1992
• "Technical Area 2 Heavy Lift Launch Vehicle Development - Concurrent Engineering Meeting # 1", J. B. McCurry, May 5 - 6, 1992
• "Advanced Transportation System Studies Heavy Lift Launch Vehicles Concept Development - Contract Kick-Off Briefing", June 17, 1992
• "ATSS TA-2 Concurrent Engineering Meeting - HLLV Design Requirements and Characteristics Results", G. F. Letchworth, August 5, 1992
• "ATSS TA-2 Quarterly Review # 1", October 2, 1992
• "Lockheed ATSS TA-2 Status", TIM # 10, November 17, 1992
• "FY 92 HLLV Development Accomplishments", J. B. McCurry
1993 Presentations:
• "ATSS TA-2 Status Review - Single Launch Lunar Launch Vehicle Configuration Options", January, 1993
• "Advanced Transportation System Studies Technical Area 2 Heavy Lift Launch Vehicle Development - Contract Status Briefing", January 27, 1993
• "50 K Launch Vehicle Definition in Support of the Access to Space Panels", J. B. McCurry, February, 1993
• "ATSS TA-2 Status Review", TIM # 11, February 11, 1993
• "Lunar Mission Architecture Utilizing Single Stage to Orbit and Heavy Lift Launch Vehicle Candidates", J. B. McCurry, February 27, 1993
• "TA-2 Support to Other NASA Projects", J. B. McCurry, March 1993
• "Alternative 50 K Vehicle Concepts - Operations Evaluation Scores", LSOC, April, 1993
• "Alternative Launch Vehicle Concepts for NASA's Access to Space Studies", K. A. Holden, K. D. Sagis, April 1993
• Untitled cost modeling methodology presentation by ECON, applied to 50-80K launch vehicle assessments, J. Skrat, ECON Inc., April 1993
• "ATSS TA-2 Contract Heavy Lift Launch Vehicle Development - Current Contract Status", J. B. McCurry, April 16, 1993
• "TA-2 Contract Status Presented to NASA Headquarters", J. B. McCurry, May 24, 1993
• "Advanced Transportation System Studies - TA-2 Contract Status", August 12, 1993

Figure 12.0-1 Major TA-2 Presentations

1994 Presentations:

- "Qualitative Assessment of Tripropellant Main Propulsion versus Bipropellant Main Propulsion for Single Stage to Orbit Vehicles", J. B. McCurry, February 7, 1994
- "Advanced Transportation System Studies TA-2 Contract Status - Assessment of Single Stage to Orbit Concepts", February 11, 1994

Figure 12.0-1 Major TA-2 Presentations (Concluded)

A copy of each of the above listed presentations is provided herein as a summarization of the significant results and conclusions that were developed over the course of the contract. The major presentations made by Pratt & Whitney regarding the identification and assessment of Russian main propulsion technologies and performance capabilities are documented as proprietary contract deliverable items, that were delivered to the TA-2 COTR. A listing of those presentation deliverables is contained in Section 13 of Volume II.

Major TA-2 Presentations Given in 1992

**LOWERING THE OPERATIONS COST
OF
NEW LAUNCH VEHICLE SYSTEMS**

ECON, INC.

FEBRUARY 11, 1992

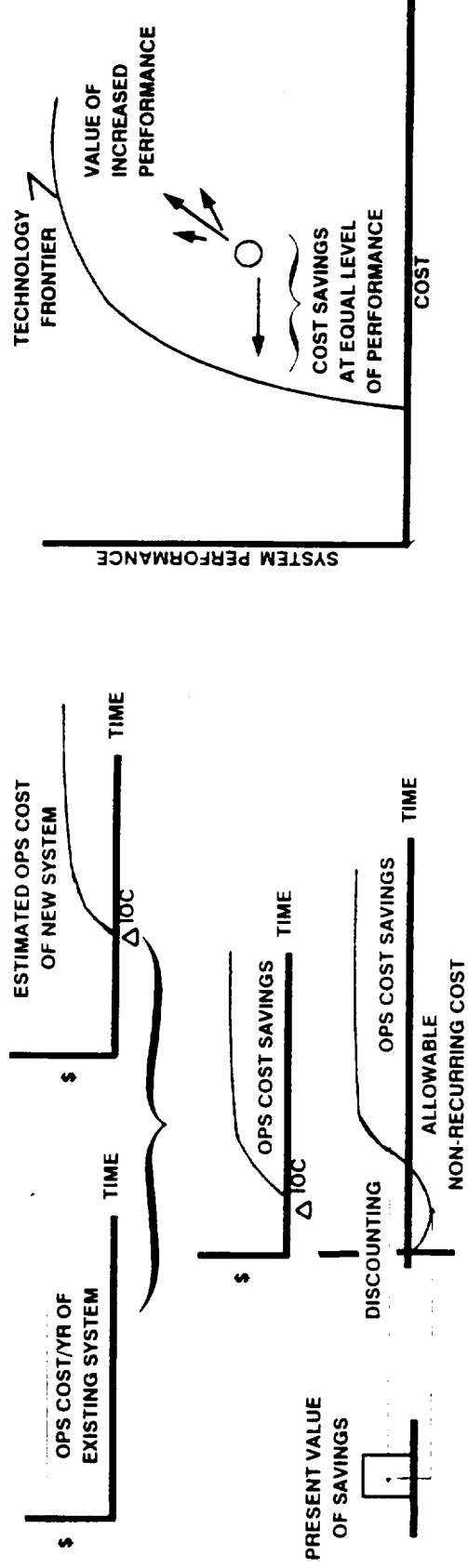


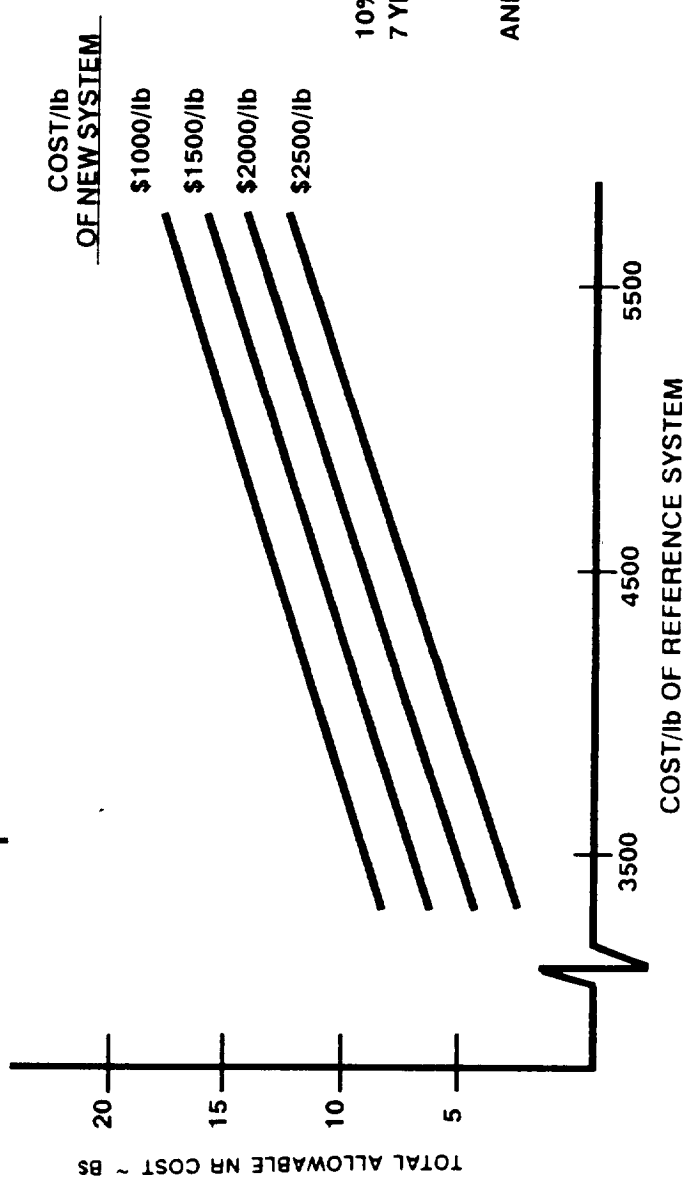
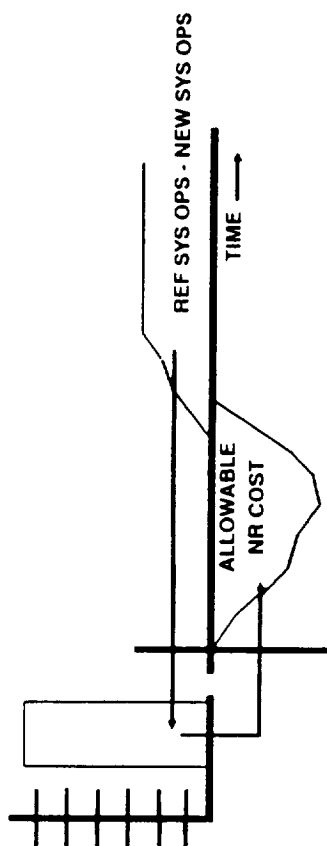
TOPICS

- GENERAL ECONOMICS OF NEW SPACE TRANSPORTATION SYSTEMS
- HOW MUCH CAN BE INVESTED IN LOWERING THE OPS COST OF LAUNCH VEHICLE SYSTEMS
- SENSITIVITIES
- THE 2 BY 2 MATRIX
- REDUCING THE OPS COST/lb FOR ELEMENTS OF THE NLS
- ESTABLISHING DESIGN/COST TRADEOFF BOUNDARIES
- SUGGESTING OPS COST REDUCTION OPTIONS REQUIRES AN INTEGRATED DESIGN/COST ASSESSMENT

GENERAL ECONOMICS OF INTRODUCING A NEW LAUNCH VEHICLE SYSTEM

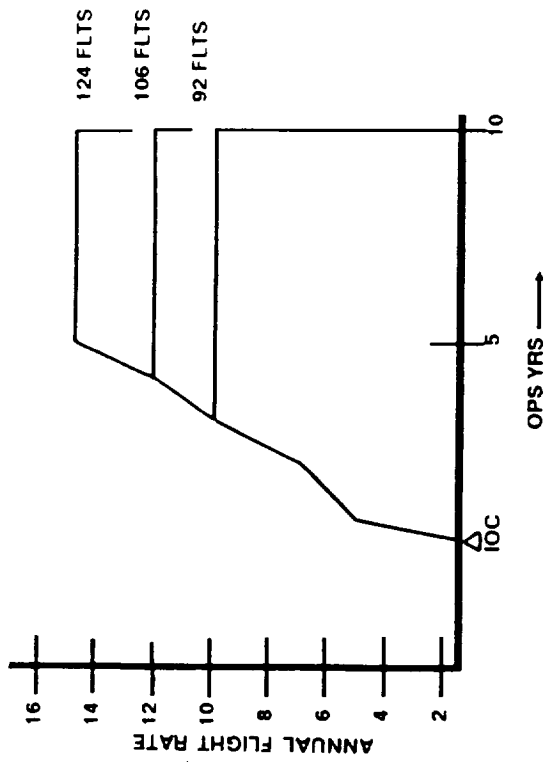
- COST AND "PERFORMANCE" OF EXISTING SYSTEM
- COST (AT EQUAL LEVEL OF "PERFORMANCE") OF NEW SYSTEM
- HOW MUCH NON-RECURRING \$ CAN BE SPENT TO IMPLEMENT OPS COST REDUCTION?



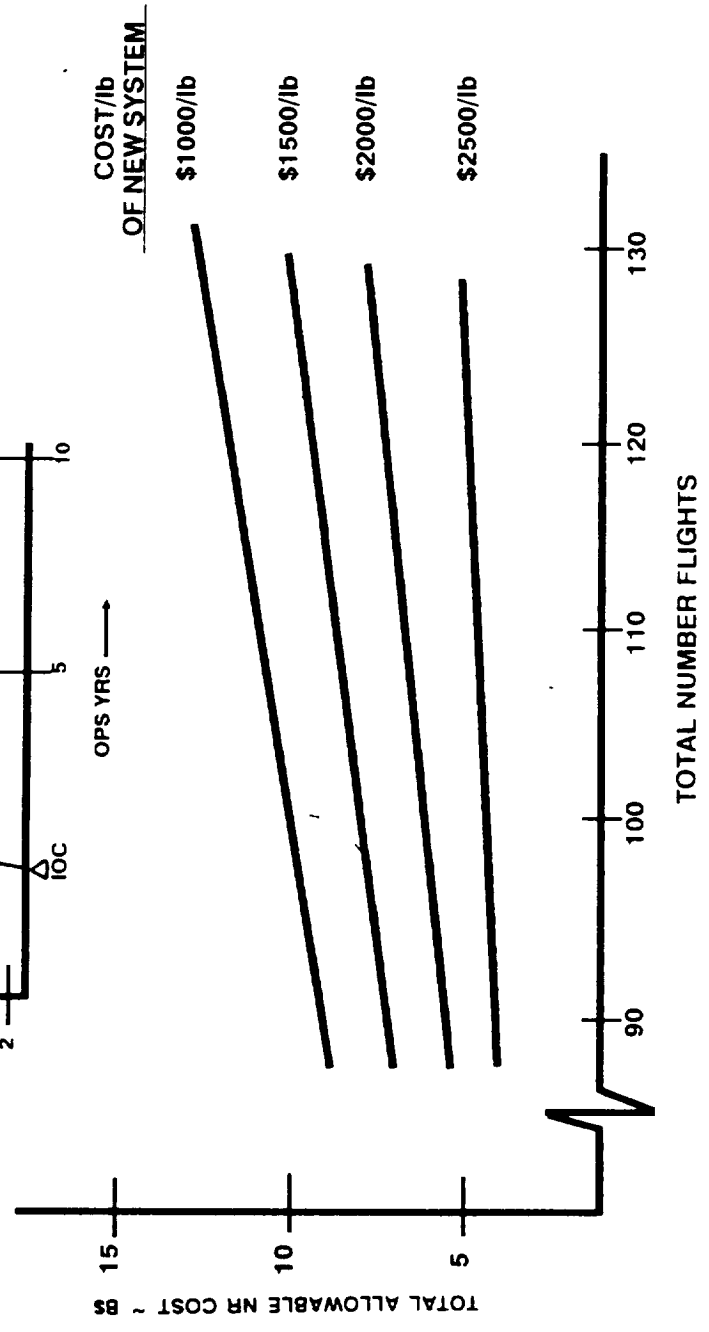


10% DISCOUNT RATE
7 YR NR PERIOD
5%, 10%, 15%, 17%, 20%, 18%, 15%

ANNUAL FLIGHT RATE
1st YEAR OPS 5 FLTS
2nd YEAR OPS 7 FLTS
3rd THRU 10th YR 10 FLTS
92 TOTAL FLIGHTS



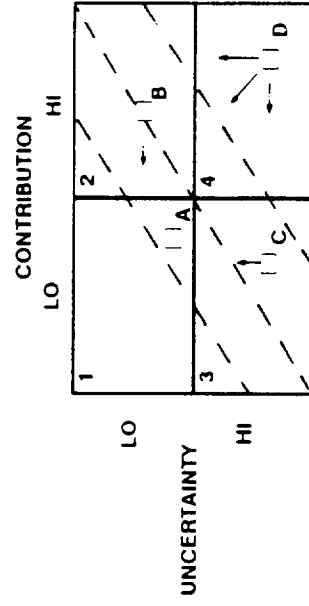
10% DISCOUNT RATE
 SAVINGS BASED ON \$3500/lb REF SYSTEM
 7 YRS NR PERIOD
 5%, 10%, 15%, 17%, 20%, 18%, 15%
 NEW SYSTEM & REF SYSTEM
 P/L CAPABILITY LEO = 55,000 lbs



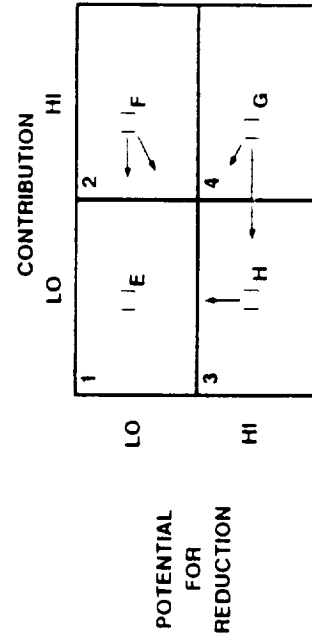
COST/lb
 OF NEW SYSTEM

THE 2 BY 2 MATRIX

- A MECHANISM FOR EFFECTIVE COST ANALYSIS FEEDBACK TO THE DESIGN PROCESS



POSITIONING, BY COST AND DESIGN ANALYSIS (JUDGMENT) OF THE ELEMENTS (1 1A...) OF A PROPOSED SYSTEM DEFINED AT ANY GIVEN TIME ~ CORRESPONDING TO A GIVEN LEVEL OF DESIGN & COST ANALYSIS



CHALLENGE THE DESIGN AND PROGRAMMATIC ASSUMPTIONS THAT CAUSE ANY ELEMENT (1 1A...) TO BE LOCATED IN QUADRANTS 2, 3 AND 4

AN EXAMPLE DRAWN FROM THE NLS COST ASSESSMENT CONDUCTED BY ECON FOR LMSC IN THE FALL OF 1991

- 1.5 STAGE NLS CONFIGURATION
- 55K lbs LEO DUE EAST
- 26 FLIGHTS
- SHARED COST OF PROPELLANT TANKS & STMEs
- MSFC/ROCKETDYNE STME COST PROJECTIONS
 - NLS OPERATIONS (EXPENDABLE HARDWARE AND LAUNCH SUPPORT) COST ELEMENTS DEFINED BY
 - WEIGHT STATEMENTS
 - LIMITED DESIGN SPECIFICITY
 - INFERRED DESIGN CHARACTERISTICS
 - ANALOGOUS ELEMENTS IN LAUNCH VEHICLE COST DATA BASE
- HOW DO THE NLS OPERATIONS COST ELEMENTS CONTRIBUTE TO TOTAL COST/lb?
- WHAT DESIGN AND PROGRAMMATIC ALTERNATIVES CONSTITUTE THE POTENTIAL FOR REDUCING THE COST OF NLS OPERATIONS - AND BY HOW MUCH?

COST/lb (26 FLTS, 55K P/L)

\$1000

\$750

\$500

\$250

0

1.5

1.4

1.3

1.2

1.1

1.0

.9

.8

.7

.6

.5

.4

.3

.2

.1

0

PROD & OPS COSTS ~ B\$

INTEGRATION

STME

OPERATIONS

F/F/D & PRES + PNEUM

AVIONICS

LOX & LH₂ TANKS

THRUST & MISC STRUCT

TPS

FAIRING

INTERTANK S/W

ROLL CNTRL & DEORB

TVC

POWER

HARDWARE -- MATERIAL, PART COUNT, TOLERANCES, MFG PROCESS, "OFF THE SHELF", TEST, SPECS, ETC.
SOFTWARE -- ADAPTATION OF EXISTING CODE, FLEXIBILITY, SLOC, LANGUAGE CODING PRODUCTIVITY, DOCUMENT., TEST, ETC.
INTEGRATION -- PARTS, TEST PROCEDURE, SPECIFICATIONS
OPERATIONS -- NO. & TYPE OF PRODUCTS & SERVICES, FACILITY AVAILABILITY, SITE INTEG & TEST, MISSION SUPPORT, ETC.
PROGRAMMATICS -- PROCUREMENT ENVIRONMENT, SPEC. LEVELS, TIMELINE, ETC.

LO

HI

ABILITY TO
DECREASE NLS OPS
ELEMENT COSTS

QUADRANT I

QUADRANT II

QUADRANT III

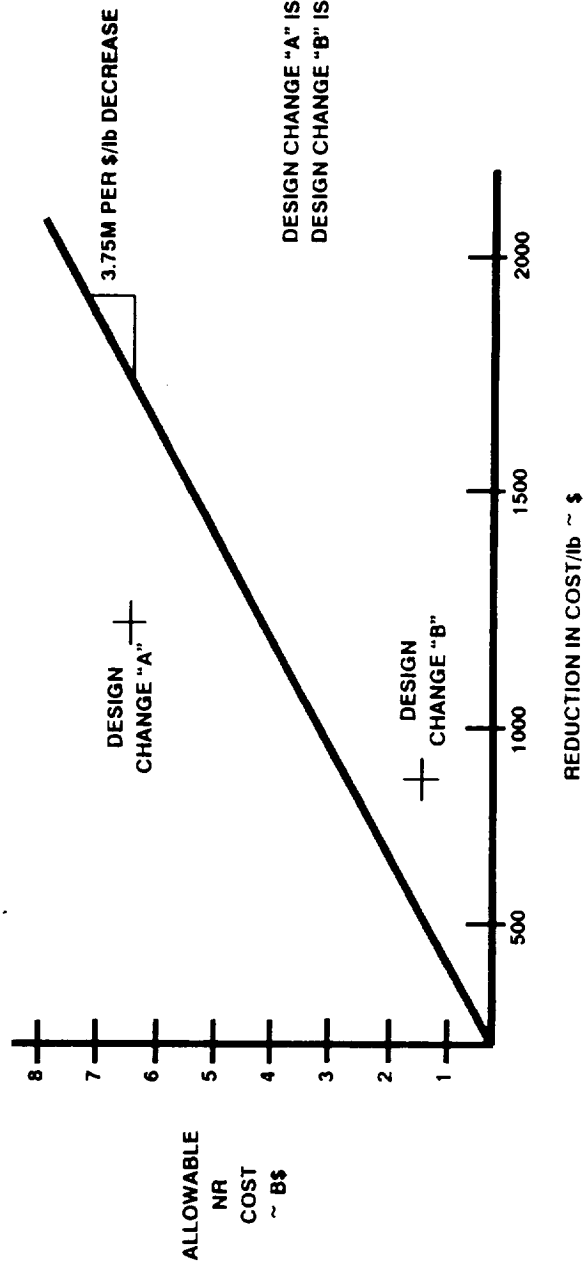
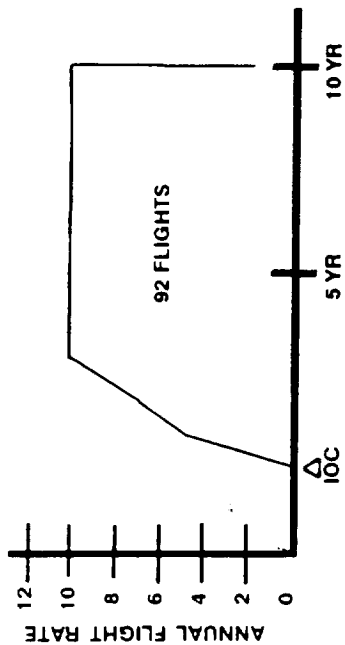
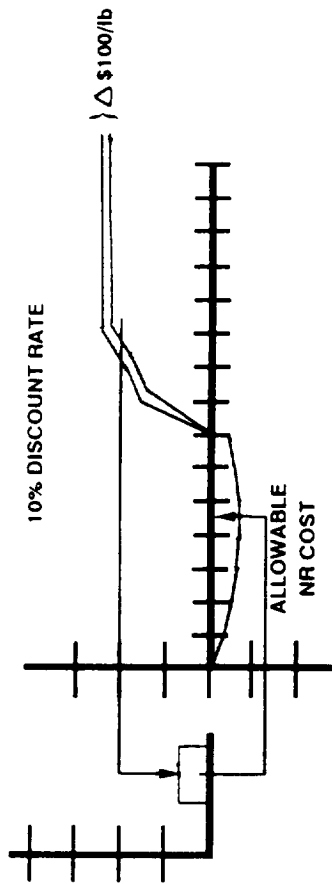
QUADRANT IV

NLS OPERATIONS
COST ELEMENTS

H/W, S/W, OPS &
PROGRAMMATIC

GIVEN THAT THERE ARE IDENTIFIABLE
DESIGN AND PROGRAMMATIC ALTERNATIVES
WHICH CAN CONTRIBUTE TO THE REDUCTION
OF OPERATIONS COSTS

HOW MUCH NON-RECURRING COST CAN BE INVESTED
IN BRINING ABOUT SUCH REDUCTIONS?



RESPONDING TO A REQUEST FOR IDEAS AS TO HOW TO REDUCE NLS OPERATIONS COSTS — REQUIRES:

1. CLEARLY STATED AND AGREED UPON COST ASSESSMENT OF ALL ELEMENTS OF NLS COSTS
2. A STRUCTURE BY WHICH TO RANK ORDER THOSE ELEMENTS THAT CAN SUBSTANTIALLY CONTRIBUTE TO COST REDUCTIONS - INCLUDING HOW SUCH REDUCTIONS COULD BE ACHIEVED
3. AN INTERACTIVE (INTEGRATED) DESIGN AND COST ASSESSMENT
 - CAN RANGE FROM EXPERT JUDGMENT TO DETAIL ANALYSES
 - MUST INCLUDE ALL OPS ELEMENTS - HARDWARE AND SOFTWARE PRODUCTION, NON-HARDWARE OPERATIONS AND PROGRAMMATIC VARIABLES

PAST ECON LAUNCH VEHICLE EXPERIENCE

SPACE TRANSPORTATION ARCHITECTURE STUDIES (STAS) - MANY VEHICLES, MANY MISSION MODELS, MANY COMBINATIONS OF MISSION MODELS AND VEHICLES - LITTLE ANALYSIS. NO TIME WAS PERMITTED TO CONDUCT PARAMETERS OF WHAT THE RESULTS MEANT OR WHAT THE DISCRIMINATORS WERE.

SHUTTLE-C - IN-LINE SYSTEM WAS GROUND RULED OUT OF EXISTENCE. SIDE MOUNT COULDN'T BE DONE FOR UNDER THE \$1B DDT&E THE GOVERNMENT WANTED. YOU ALSO RAN OUT OF "USED" SSME'S PRETTY QUICKLY, WHICH MADE EACH FLIGHT VERY EXPENSIVE.

ADVANCED LAUNCH SYSTEM (ALS) - COST OBJECTIVES WERE UNREALISTICALLY GROUND RULED BY CONGRESS. LOTS OF LIP SERVICE TO "NEW WAYS OF DOING BUSINESS". GOVERNMENT EVENTUALLY FORCED CONVERGENCE OF ANSWER. DESIGN AND COST NOT EFFECTIVELY INTEGRATED.

NATIONAL LAUNCH SYSTEM (NLS) - BUILT ON THE SUPPOSITION THAT A GOVERNMENT DEVELOPED COMMON ENGINE AND A SEMI-COMMON CORE MAKE FOR A COST EFFECTIVE LAUNCH VEHICLE FAMILY - BUT A \$12B DEVELOPMENT PROGRAM FOR NLS DOESN'T BUY A COST-PER-FLIGHT LESS THAN A TITAN OR ATLAS.

ECON'S LAUNCH VEHICLE TRADE EXPERIENCE

■ System Trades

- Degree of Reusability
- Vehicle Payload Capability
- Flight Performance - Loads & Weights
- Subsystem Performance/Requirements
- Engine Out vs. High Reliability Engines
- Test Program Structure
- Producability/Manufacturing Trades

■ Design Trades

- Struct Design Philosophies & Mat'l Alt'tives
- Thermal Management/Handling Options
- Primary & Secondary Propulsion Alternatives
 - Propellant Options
 - Engine Design Options
 - Thrust Vectoring Options
- Avionics - Autonomy & Redundancy
- Leverage or Suitability of SOTA Technology
- "ilities" Trades
 - Availability
 - Reliability
 - Resiliency
 - Maintainability
 - Affordability

■ Operations Trades

- Ground Processing Options
 - Horizontal vs. Vertical Assembly
 - Integrate-Transfer-Launch vs. Assemble at Pad
- Launch Site Alternative Evaluation
 - Existing Sites vs. All-azimuth site
 - New facilities/GSE vs. Shared facilities
- Vehicle HMS/BITE vs. Ground Processing Testing/Checkout
- Mission Planning & Mission Design Alternatives
- Guidance Dataloads
 - Adaptive Guidance
 - Day-of-Launch Dataloads
 - Standard Dataloads
 - Flight Specific Dataloads
- Logistics Philosophy Alternatives
 - Sparing
 - Depot/Backshop

WHY LAUNCH VEHICLES COST WHAT THEY DO

DDT&E

- Long development durations
- Uncertain Budgets
- Very little test hardware/research

PRODUCTION

- Low Production Rates
- Unique Parts
- Strict Tolerances
- Tight Margins - Shave weight
- Paper Trail
- Expendable
- High Reliability Requirements - Space Qual Parts
- Exotic Materials

OPERATIONS

- Standing Army - Need a Charge Number
- Serial Processing
- Uncertain Budgets
- No competition/no incentive to cut costs & streamline
- Low Launch Rates - Amortize Large Fixed Costs
- Old Infrastructure to Support Processing and Launch
- Unique Missions
 - Orbital (Mission) Profiles
 - Unique Interfaces
 - Payload Access on Pad

OTHER RANDOM THOUGHTS

The predominate objective of NASA is research and development. When applied to launch vehicles this R&D paradigm results in long development cycles which correlates directly to large bucks. SEI may want a relatively inexpensive transport but the transport designers want a thoroughly tested bird. A possible design criteria is - what design can be flown the quickest.

Any analyses applied to the process of defining an effective and efficient launch vehicle design requires the application of tools which evolve along with the concept - that is the analysis must take into consideration an increasing amount of detail as the design matures. The challenge is to match the tool (level of detail) to the status of the design information. This is true whether we are talking about trajectory design, structural loads, reliability or cost. The validity of any answer concerning launch vehicle design, at any one point in time, depends upon the degree to which the analytical tools have captured the appropriate level of detail

From a cost perspective new work needs to be done in support of the SEI HLLV designs to adequately capture "new ways of doing business", inheritance of Saturn V designs, the large size of certain components and the application of Russian hardware or technology

It would appear that with all the Red and Blue Team activity at NASA that there is a great diversity of cost data being brought forth concerning competing designs. There is a need to generate comparable costs for all competing designs.

SPECIFIC LESSONS LEARNED
In Economics of Launch Vehicles

PROGRAMMATIC EFFECTS ON ECONOMICS

Large Lot Buys Are Key to Expendable Hardware:

- * At All Costs Avoid One- or Two-Vehicle Buys

Bulk Cargo Logistics Are a Factor:

- * Different from Manned/High Value Payloads

Schedule Affects Cost:

- * Long Programs Are Always Expensive

Cost of Failure Should be Primary Trade:

- * Loss of Vehicle and Payload
- * Loss of Revenue Opportunity
- * Traded Against Cost of Reliability Increase

Government Oversight Also Involves a Trade:

- * Presumption Has Been That Oversight Reduces Risk
- * In Commercial Aircraft Industry, Manufacturer Bears Risk

DESIGN EFFECTS ON ECONOMICS

Weight Can be Traded for Cost:

- * Example is Ariane Propellant Tanks

Avionics and Liquid Engines Drive Expendable Vehicle Costs

Design Sophistication Penalizes Expendable Hardware:

- * Example is STME Computer

FIRST LUNAR OUTPOST
HEAVY LIFT LAUNCH VEHICLE DEVELOPMENT

NLS-DERIVED CONFIGURATION ASSESSMENT

PERFORMED BY LOCKHEED
FOR THE
MARSHALL SPACE FLIGHT CENTER

Work Performed During February-March 1992

FIRST LUNAR OUTPOST NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

INTRODUCTION

Lockheed was tasked by the Space Transportation and Exploration Office (PT01) of the Marshall Space Flight Center (MSFC) to develop and assess candidate Heavy Lift Launch Vehicle (HLLV) configurations derived from the National Launch System (NLS). The configurations were to be sized to meet the Single Launch lunar mission scenario. NLS-derived launch vehicles were sized and their nominal ascent performance assessed with the design goal of providing a 93 metric ton payload capability after completion of the Trans-Lunar Injection (TLI) burn. Lockheed's design philosophy was to maximize vehicle element commonality, minimize vehicle dry weight, minimize structural and main propulsion subsystem design complexity, and to allow for performance growth options; all while adhering to facility constraints at the Kennedy Space Center (KSC). A preliminary assessment of launch site facility issues and requirements was also performed, with an emphasis on identifying first order vehicle design drivers. The ground processing facility physical limitations ended up imposing a fundamental limitation on the design of the vehicle by limiting the vehicle's length in order to utilize the current Vehicle Assembly Building (VAB).

FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT
SINGLE LAUNCH DESIGN OPTIONS

The two most overriding groundrules that were received from MSFC/PT01 were to have common core vehicle and booster tank diameters, and to limit the length of the core vehicle liquid hydrogen (LH2) tank to no more than 15 feet (4.6 meters) greater than that for the Space Shuttle External Tank (ET). The intent of those groundrules was to limit design, development, test, and evaluation (DDT&E) costs. The 15 foot tank extension limit was based upon minimizing the redesign cost impact to an LH2 tank static load cell at the Michoud Assembly Facility (MAF). Brent Clayborn, lead for NLS cost analysis for Martin Marietta Manned Space Systems (MMMSS) Division (504-257-2091) indicated to Lockheed that 10 foot extensions to the ET LH2 tank could be accommodated without any design impacts to the load cell. A 15-18 foot (4.6-5.5 meter) extension could be accommodated with minimum to moderate cell modifications. Extensions greater than 18 feet would result in substantial cell modifications, and equate to the cost of a new check-out cell.

A similarly imposing PT-01 groundrule was to exclusively use F-1A engines on the boosters. Terry Murphy, Manager of F-1 Program Development for Rocketdyne (818-718-4851), was contacted to identify absolute performance limitations of the proposed F-1A engine. He stated that two 100 percent rated power level (RPL) thrust levels were possible, based upon actual full-scale testing, 1.8 E6 lbf (8.0 E6 N) to 2.0 E6 lbf (8.9 E6 N) sea level thrust. A range of minimum throttle settings was also possible between 65 and 75 percent RPL.

By freezing the core and booster tank diameters to that of the Space Shuttle ET (27.6 feet; 8.4 meters), and by limiting the length of the core LH2 tank, the booster attach point locations became defined a priori. Limiting the attach points therefore limited the booster propellant loading. The remaining open design parameters became the F-1A thrust level, number of booster engines, number of boosters, TLI stage propellant loading, type of TLI stage engine, and number of TLI stage engines. Lockheed identified alternate design methods that would increase vehicle performance while maintaining the attach point and LH2 tank length constraints, such as a second LH2 tank for the core vehicle, LOX tank aft for the boosters, common propellant tank bulkheads for the boosters and/or TLI stage, and use of Aluminum-Lithium. While these methods were quantitatively assessed by Lockheed, the resulting configurations were felt by MSFC/PT01 to be programmatically unacceptable at this time.

A variety of design options were assessed for each of the NLS-derived configuration elements, as well as for manufacturing methods and vehicle performance assessments. Lockheed was also asked to assess three-stage core vehicle concepts, which contained the basic NLS core vehicle, a new LOX/LH2 second stage, and a LOX/LH2 TLI stage, in addition to two LOX/RP-1 boosters. This configuration was to identify any performance

FIRST LUNAR OUTPOST NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

SINGLE LAUNCH DESIGN OPTIONS (Continued)

payoffs for replacing two parallel-burn boosters with one series-burn upper stage.

Shroud:

The biconic shroud initially baselined by MSFC/PD24 was utilized, with the associated mass properties. NLS forebody aerodynamics representing a Titan IV biconic shroud were also used.

TLI Stage:

Three engine types were assessed: J-2S, Space Transportation Main Engine (STME), and Space Shuttle Main Engine (SSME). Two-engine combinations were sized for use of STMEs or SSMEs, while use of six or eight J-2Ss were sized. Due to anticipated main propulsion feed system and thrust structure complexities associated with engine clusters of greater than four, the J-2S was dropped as a viable candidate. The mass properties were determined through the use of a mass fraction derived from the S-IVB stage, that was a function of the TLI stage propellant load. It was recognized that the S-IVB used common bulkheads for the propellant tanks, while the NLS-derived TLI stage did not. Thus the mass fraction was slightly optimistic, although a 10 percent inert mass margin was also accounted for. A bottoms-up mass properties assessment based upon those used by NLS was to be performed at a later date. Two propellant loads were sized: one for a minimum amount based upon a dome-to-dome LOX tank (ET diameter), which gave 590,000 lbm of propellant, and a performance-optimal propellant loading, which gave 760,000 lbm of propellant when adhering to the VAB high bay vertical clearance limit.

Second Stage:

Two engine types were assessed: SSMEs and STMEs. Two-engine combinations were sized. The mass properties were determined through the use of a mass fraction derived from the S-IVB stage, that was a function of the TLI stage propellant load. Three propellant loads were sized, corresponding to 5, 10, and 15 foot extensions to the LH2 tank.

NLS Core:

The NLS reference HLLV core vehicle was used, which contained four STMEs. NLS Cycle 0 mass properties were used and three propellant loads were sized, corresponding to 5, 10, and 15 foot extensions to the LH2 tank.

FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT
SINGLE LAUNCH DESIGN OPTIONS (Concluded)

Boosters:

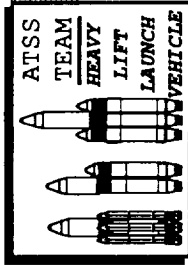
The F-1A engine was used at two maximum thrust levels: 1.8 E6 lbf (8.0 E6 N) and 2.0 E6 lbf (8.9 E6 N) sea level thrust. Three and four-engine combinations were sized. The propellant loading was sized based upon the location of the core vehicle's attach struts, which was a function of the LH2 tank stretch quantity. The mass properties were determined through the use of a mass fraction derived from the S-IC stage, that was a function of the TLI stage propellant load. Four different booster engine layouts were assessed for controllability, structural, and plume heating issues.

Manufacturing Methods:

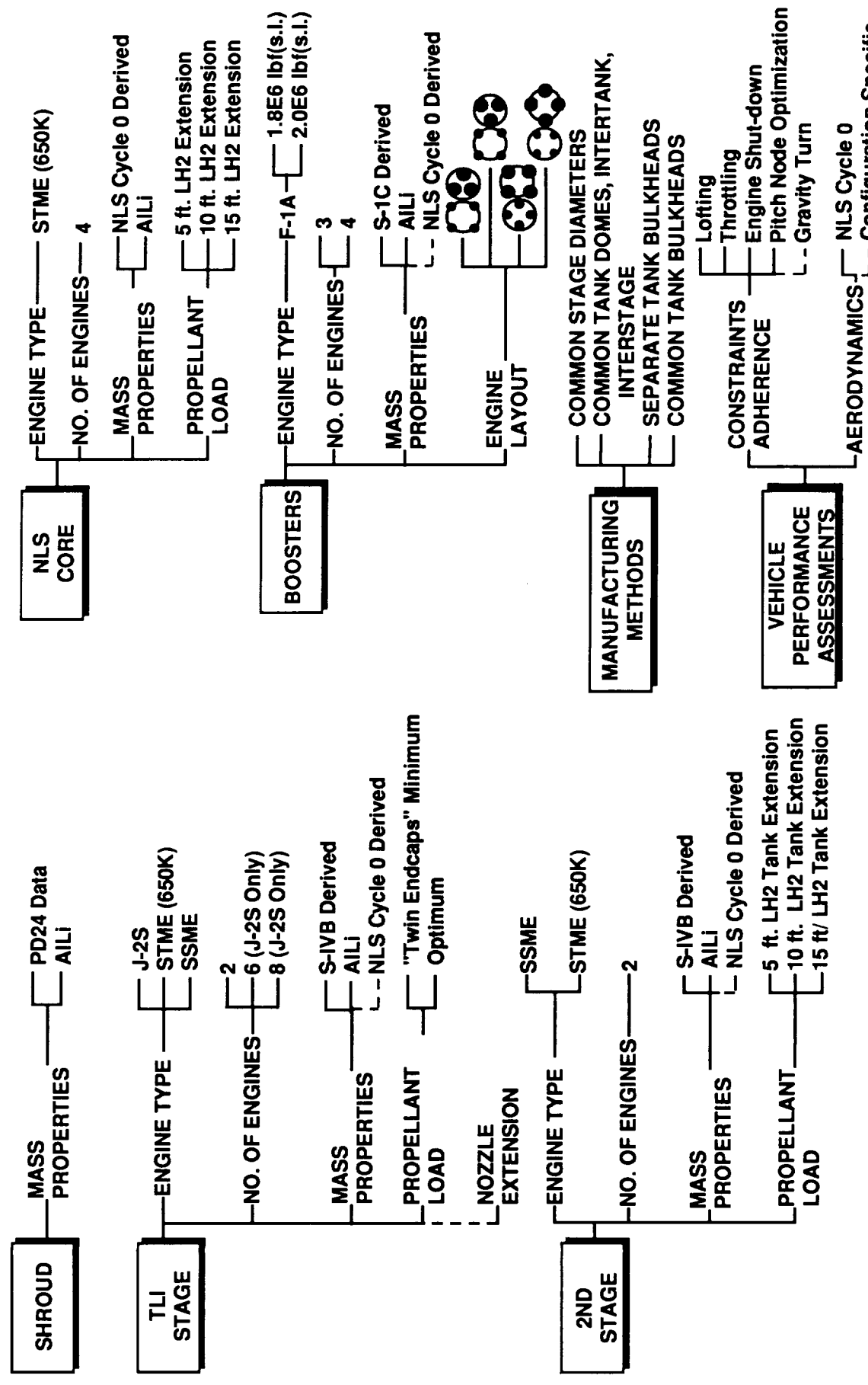
In order to minimize manufacturing and tooling costs, each of the vehicle elements utilized common stage diameters, common tank domes, common intertanks, common interstages (where applicable), and separate propellant tank bulkheads. The relative size benefit (and thus performance benefit) of utilizing common propellant tank bulkheads was assessed for each of the stage elements.

Vehicle Performance Assessments:

The Simulation and Optimization of Rocket Trajectories (SORT) three-degree-of-freedom simulation and optimization tool was used to assess the nominal ascent performance of the candidate vehicle configurations and to help in refining vehicle sizing. The ascent trajectories were optimized subject to dynamic pressure and thrust acceleration constraints. The dynamic pressure constraint was adhered to during ascent via two methods: trajectory lofting and stage engine throttling. The acceleration constrain was adhered to via two methods: stage engine throttling and engine shut-down. NLS Cycle 0 aerodynamics (forebody and base effects) were also used.



SINGLE-LAUNCH DESIGN OPTIONS



NOTE: Dashed line indicates not yet assessed.

FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT
STACK LIFT-OFF THRUST-TO-WEIGHT RATIO

An early question that needed to be answered in sizing the NLS-derived configurations was how many engines to have on each booster. An assessment of lift-off thrust-to-weight ratios was performed as a function on number of boosters, number of booster engines, booster engine thrust level, booster propellant load, and core stage propellant load. The candidate vehicle configurations consisted of a core stage, a trans-lunar injection (TLI) stage and either two or four boosters strapped onto the side of the core stage. Two propellant loads were used on the core and booster stages. Two, three or four F-1As were used on each booster. The F-1A engines were run at two sea level thrust values. A lift-off thrust-to-weight ratio of 1.25 was considered to be the minimum acceptable value. Engine-out capability at lift-off was groundruled to not be a requirement.

The conclusion reached was that vehicle configurations with two F-1As per booster do not have sufficient thrust to be viable designs. Therefore, vehicle configurations with three and four F-1A per booster were used in further analysis.

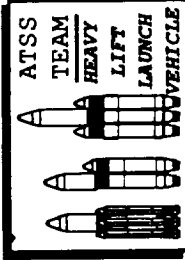
This analysis used the following vehicle configuration assumptions:

- The payload mass was 205,000 lbm (92,996 kg).
- The payload shroud mass was 35,500 lbm (16,103 kg) and was jettisoned at a geodetic altitude of 400,000 ft (121,920 m).
- The TLI stage burnout mass was 78,900 lbm (35,788 kg) and contained 590,000 lbm (267,619 kg) of propellant.
- The nominal core stage was based on the Shuttle ET with a five foot extension in the ET's liquid LH2 tank. The nominal core burn-out mass was 187,800 lbm (85,185 kg) and contained 1,690,000 lbm (766,571 kg) of propellant.
- The stretched core stage was based on the ET with a fifteen foot extension in the ET's liquid LH2 tank. The stretched core burn-out mass was 206,400 lbm (93,621 kg) and contained 1,858,000 lbm (842,775 kg) of propellant.

FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

STACK LIFT-OFF THRUST-TO-WEIGHT RATIO (Concluded)

- Both core stage configurations had four STME engines. These engines had a sea level thrust of 551,000 lbf (2,450,970 N) and a vacuum thrust of 650,000 lbf (2,891,344 N).
- The nominal boosters contained 2,900,000 lbm (1,315,418 kg) of propellant. Nominal boosters with two, three and four F-1A engines per booster had a burn-out weight of 180,400 lbm (81,828 kg), 207,900 lbm (94,302 kg) and 235,500 lbm (106,821 kg) respectively.
- The stretched boosters contained 3,260,000 lbm (1,478,711 kg) of propellant. Stretched boosters with two, three and four F-1A engines per booster had a burn-out weight of 195,700 lbm (88,768 kg), 223,300 lbm (101,287 kg) and 250,000 lbm (113,398 kg) respectively.
- The F-1A engines had a sea level thrust of 1,800,000 lbf (8,006,799 N) or 2,000,000 lbf (8,896,443 N).



STACK LIFT-OFF THRUST-TO-WEIGHT RATIO

F-1As @ 1.8E06 lbf (sea level)	Number of F-1As on a Booster		
	2 F-1As	3 F-1As	4 F-1As
Nominal Core and 2 Boosters	1.051	1.444	1.833
Stretched Core and 2 Boosters	0.951	1.415	1.784
Nominal Core and 4 Boosters	1.099	1.564	2.023
Stretched Core and 4 Boosters	0.989	1.408	1.822

F-1As @ 2.0E06 lbf (sea level)	Number of F-1As on a Booster		
	2 F-1As	3 F-1As	4 F-1As
Nominal Core and 2 Boosters	1.140	1.558	2.010
Stretched Core and 2 Boosters	1.032	1.547	1.956
Nominal Core and 4 Boosters	1.205	1.722	2.231
Stretched Core and 4 Boosters	1.084	1.550	2.010

Rule of Thumb: Minimum nominal thrust-to-weight @ lift-off ≥ 1.25

Conclusion: Vehicle configurations with 2 F-1As per booster do not have sufficient thrust to be viable designs.

FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

S-1C DERIVED MASS FRACTIONS AS A FUNCTION OF BOOSTER PROPELLANT LOAD

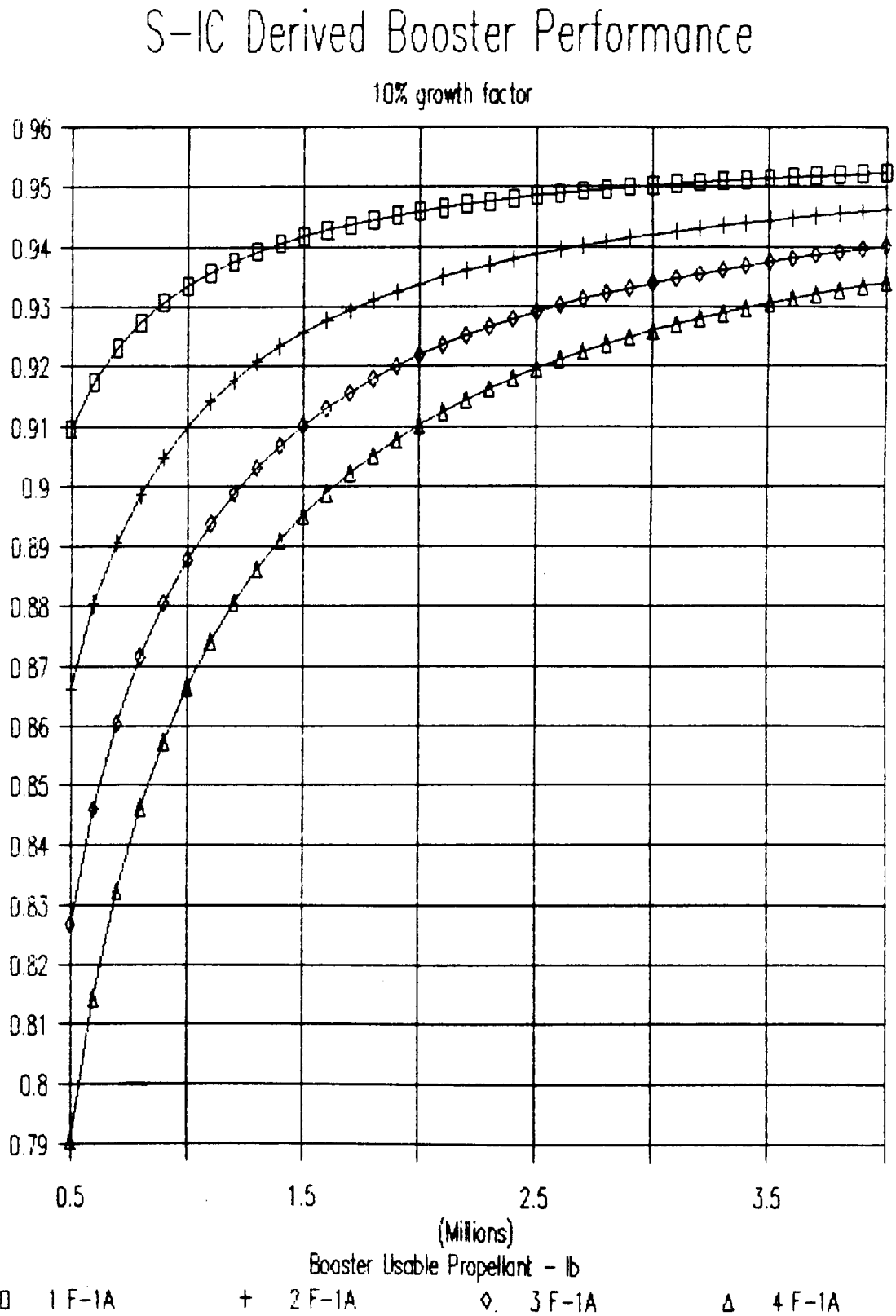
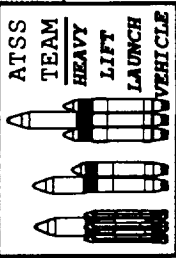
During the sizing of the NLS-derived configurations, it was necessary to estimate the burn-out weight of the boosters to calculate the performance of the vehicle configuration. Since propellant mass fractions change slowly, one of the best ways to do this was to use a booster propellant mass fraction. An iterative approach was used in which a booster propellant load was selected. The resulting booster propellant mass fraction was looked up and the vehicle configuration performance was calculated. Based on this vehicle performance, a new propellant load was selected.

The Saturn S-1C stage propellant mass fraction was used as the basis for calculating the NLS-derived configuration booster propellant mass fraction. This decision was made on the basis that both stages used the same propellant combination (LOX and RP-1) and are of similar size. The propellant mass fraction was adjusted to allow for changes in the number of engines on the booster and changes in the booster propellant load.

The booster stage burn-out mass was comprised of the sum of the engine, thrust structure, tankage, growth factor and residual propellant masses. The engine mass was 19,000 lbm (8,618 kg) per F-1A engine. The thrust structure mass was 0.00300 lbm/lbf (0.000306 kg/N) times the engine vacuum thrust. The tankage mass was derived from the Saturn S-1C stage dry mass. The mass of the S-1C tankage was estimated by subtracting the mass of the F-1 engines and thrust structure from the S-1C dry mass. The resulting tankage mass was divided by the S-1C propellant load to find a unit tankage mass of 0.037255 lbm/lbm (0.03755 kg/kg) of propellant. A mass growth margin of ten percent was added to the above terms. The propellant residual mass was 0.0500 lbm/lbm (0.0227 kg/kg) times the propellant mass raised to an exponent of 0.79. The summation of the above terms equaled the stage burn-out mass.

The stage propellant mass fraction was found by dividing the stage propellant mass by the sum of the propellant mass and the burn-out mass. The booster propellant mass fraction improved as the propellant load increased and dropped as additional engines were added to the booster.

S-1C DERIVED MASS FRACTIONS AS A FUNCTION OF BOOSTER PROPELLANT LOAD



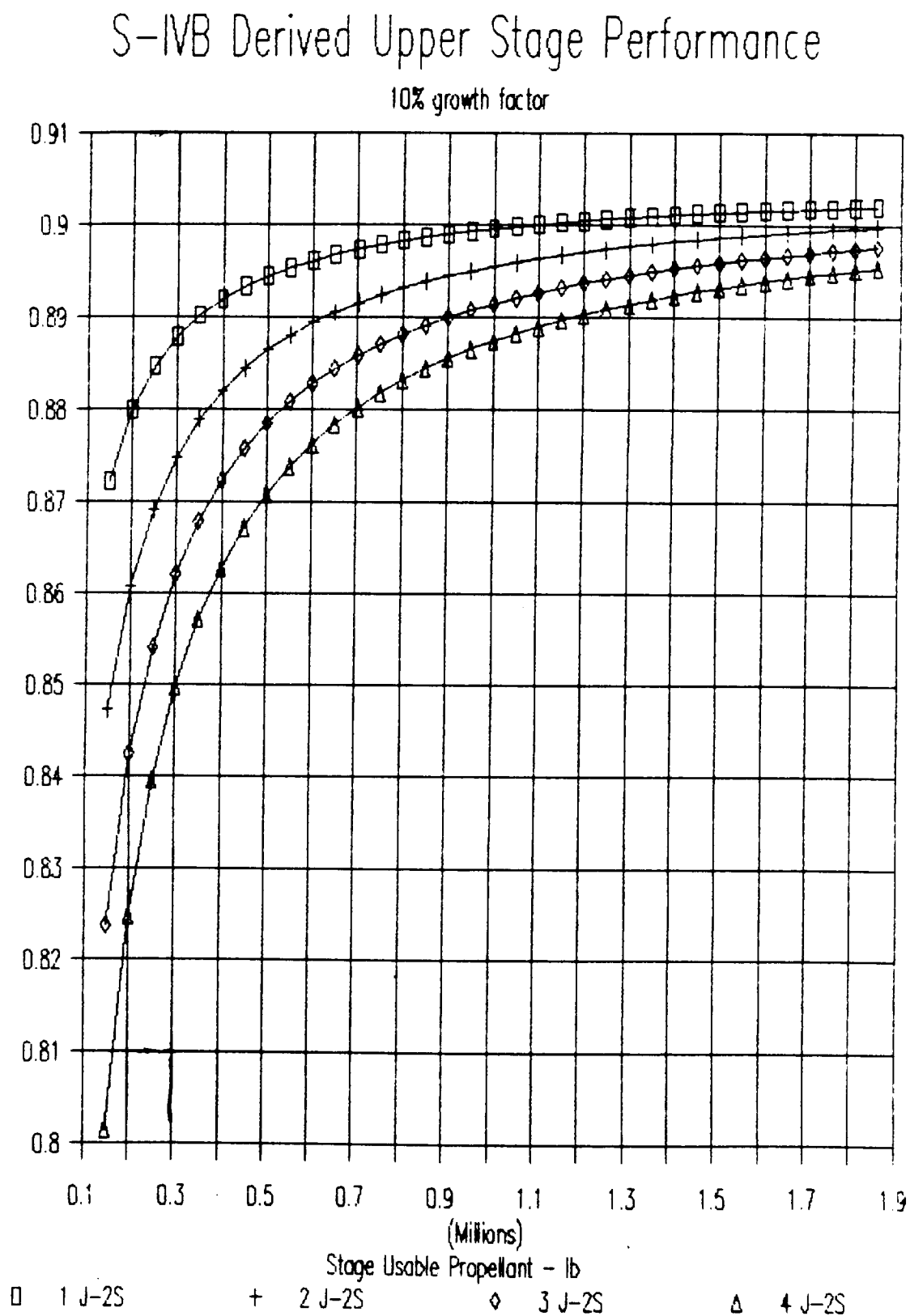
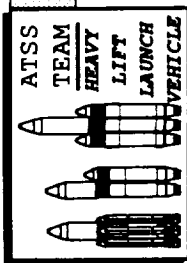
FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT
S-IVB DERIVED MASS FRACTIONS AS A FUNCTION OF STAGE PROPELLANT LOAD

The Saturn S-IVB stage propellant mass fraction was used as the basis of calculating the NLS-derived configuration trans-lunar injection (TLI) stage propellant mass fraction. This decision was made on the basis that both stages used the same propellant combination (LOX and LH2) and are of similar size. The propellant mass fraction was adjusted to allow for changes in the number of engines on the TLI stage and changes in the TLI stage propellant load.

The TLI stage burn-out mass was comprised of the sum of the engine, thrust structure, tankage, growth factor and residual weights. The engine mass was 3,800 lbm (1,724 kg) per J-2S engine. The thrust structure mass was 0.00300 lbm/lbf (0.000306 kg/N) times the engine vacuum thrust. The tankage mass was derived from the Saturn S-IVB stage dry mass. The mass of the S-IVB tankage was estimated by subtracting the mass of the J-2 engines and thrust structure from the S-IVB dry mass. The resulting tankage mass was divided by the S-IVB propellant load to find a unit tankage mass of 0.0936087 lbm /lbm (0.0936087 kg/kg) of propellant. The avionics mass was 800 lbm (363 kg). A mass growth margin of ten percent was added to the above terms. The residual mass was 0.0500 lbm/lbm (0.0227 kg/kg) times the propellant mass raised to an exponent of 0.79. The summation of the above terms was the stage burn-out mass.

The stage propellant mass fraction was found by dividing the stage propellant mass by the sum of the propellant mass and the burn-out mass. The TLI stage propellant mass fraction improved as the propellant load increased and dropped as additional engines were added to the TLI stage.

S-IVB DERIVED MASS FRACTIONS AS A FUNCTION OF STAGE PROPELLANT LOAD



FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

S-IVB DERIVED MASS FRACTIONS AS A FUNCTION OF STAGE PROPELLANT LOAD (Continued)

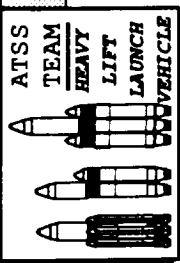
The Saturn S-IVB stage propellant mass fraction was used as the basis of calculating the NLS-derived configuration trans-lunar injection (TLI) stage propellant mass fraction. This decision was made on the basis that both stages used the same propellant combination (LOX and LH2) and are of similar size. This propellant mass fraction was adjusted to allow for changes in the number of engines on the TLI stage and changes in the TLI stage propellant load.

The TLI stage burn-out mass was the sum of the engine, thrust structure, tankage, growth factor and residual propellant mass. The engine mass was 6,990 lbm (3,171 kg) per SSME engine. The thrust structure mass was 0.00300 lbm/lbf (0.000306 kg/N) times the engine vacuum thrust. The tankage mass was derived from the Saturn S-IVB stage dry mass. The mass of the S-IVB tankage was estimated by subtracting the mass of the J-2 engines and thrust structure from the S-IVB dry mass. The resulting tankage mass was divided by the S-IVB propellant load to find a unit tankage mass of 0.0936087 lbm /lbm (0.0936087 kg/kg) of propellant. The unit tankage mass was changed to 0.08891 lbm/lbm (0.08891 kg/kg) of propellant because of the change in propellant mixture ratio of 5.50 for the J-2 engine to 6.0 for the SSME engine. The avionics mass was 800 lbm (363 kg). A mass growth margin of ten percent was added to the above terms. The propellant residual mass was 0.0500 lbm/lbm (0.0227 kg/kg) times the propellant mass raised to an exponent of 0.79. The summation of the above terms was the stage burn-out mass.

The stage propellant mass fraction was found by dividing the stage propellant mass by the sum of the propellant mass and the burn-out mass. The TLI stage propellant mass fraction improved as the propellant load increased and dropped as additional engines were added to the TLI stage.

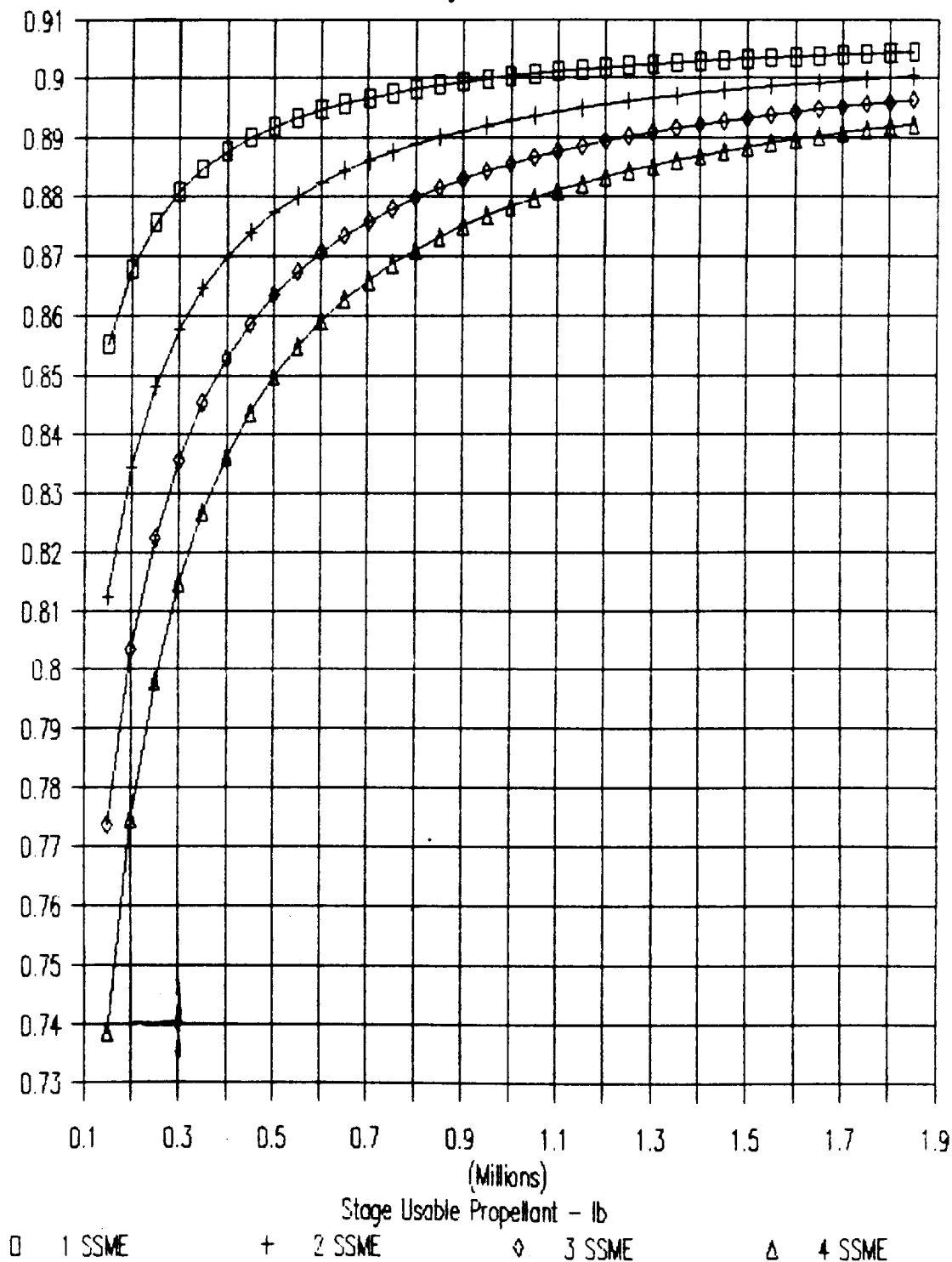
S-IVB DERIVED MASS FRACTIONS

AS A FUNCTION OF STAGE PROPELLANT LOAD (Continued)



S-IVB Derived Upper Stage Performance

10% growth factor



FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

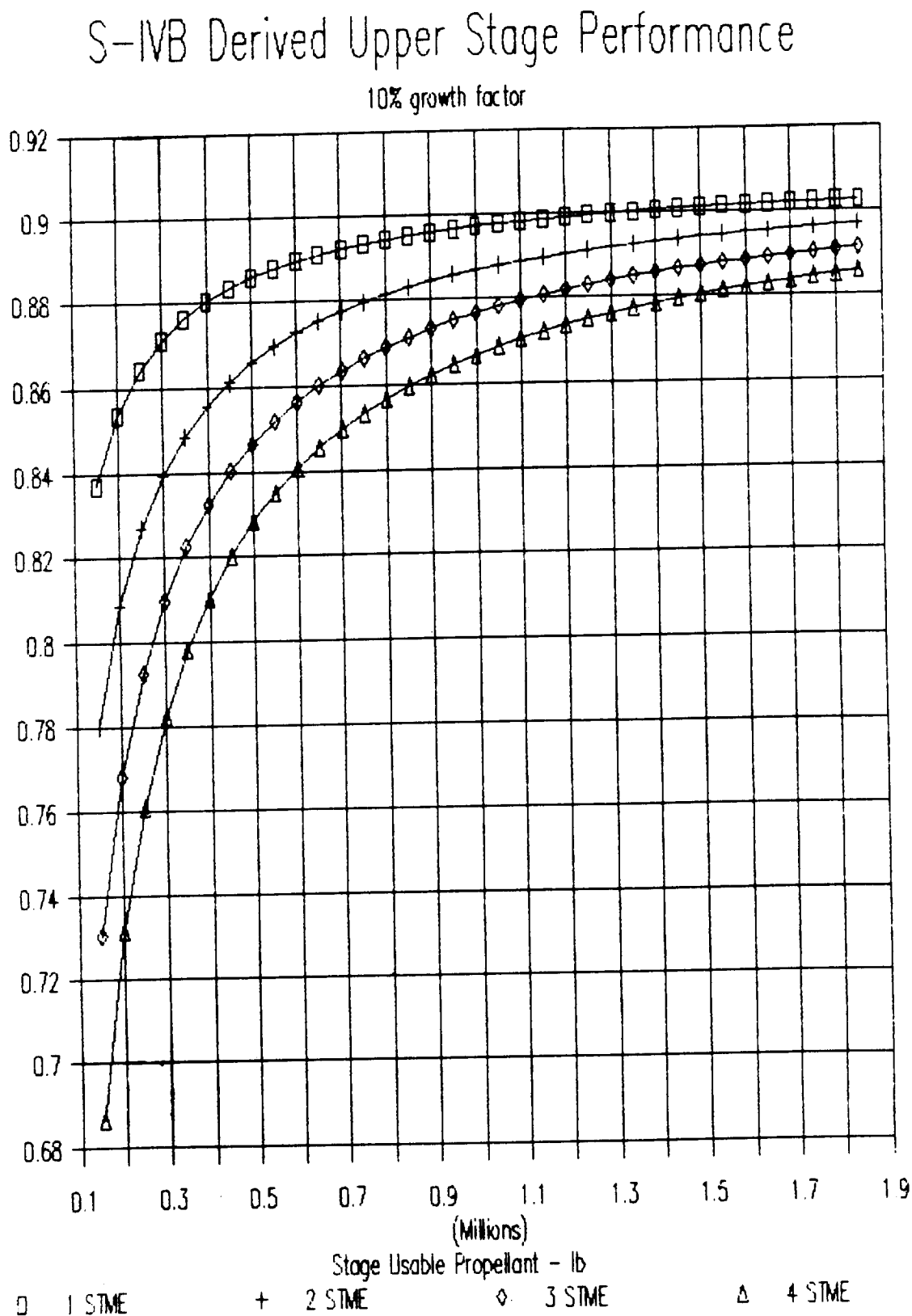
S-IVB DERIVED MASS FRACTIONS AS A FUNCTION OF STAGE PROPELLANT LOAD (Continued)

The Saturn S-IVB stage propellant mass fraction was used as the basis of calculating the NLS-derived configuration trans-lunar injection (TLI) stage propellant mass fraction. This decision was made on the basis that both stages used the same propellant combination (LOX and LH2) and are of similar size. This propellant mass fraction was adjusted to allow for changes in the number of engines on the TLI stage and changes in the TLI stage propellant load.

The TLI stage burn-out mass was the sum of the engine, thrust structure, tankage, growth factor and residual propellant mass. The engine mass was 9,930 lbm (4,504 kg) per STME engine. The thrust structure mass was 0.00300 lbm/lbf (0.000306 kg/N) times the engine vacuum thrust. The tankage mass was derived from the Saturn S-IVB stage dry mass. The mass of the S-IVB tankage was estimated by subtracting the mass of the J-2 engines and thrust structure from the S-IVB dry mass. The resulting tankage mass was divided by the S-IVB propellant load to find a unit tankage mass of 0.0936087 lbm/lbm (0.0936087 kg/kg) of propellant. The unit tankage mass was changed to 0.08891 lbm/lbm (0.08891 kg/kg) of propellant because of the change in propellant mixture ratio of 5.50 for the J-2 engine to 6.0 for the STME engine. The avionics mass was 800 lbm (363 kg). A mass growth margin of ten percent was added to the above terms. The propellant residual mass was 0.0500 lbm/lbm (0.0227 kg/kg) times the propellant mass raised to an exponent of 0.79. The summation of the above terms was the stage burn-out mass.

The stage propellant mass fraction was found by dividing the stage propellant mass by the sum of the propellant mass and the burn-out mass. The TLI stage propellant mass fraction improved as the propellant load increased and dropped as additional engines were added to the TLI stage.

S-IVB DERIVED MASS FRACTIONS AS A FUNCTION OF STAGE PROPELLANT LOAD (Continued)



FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

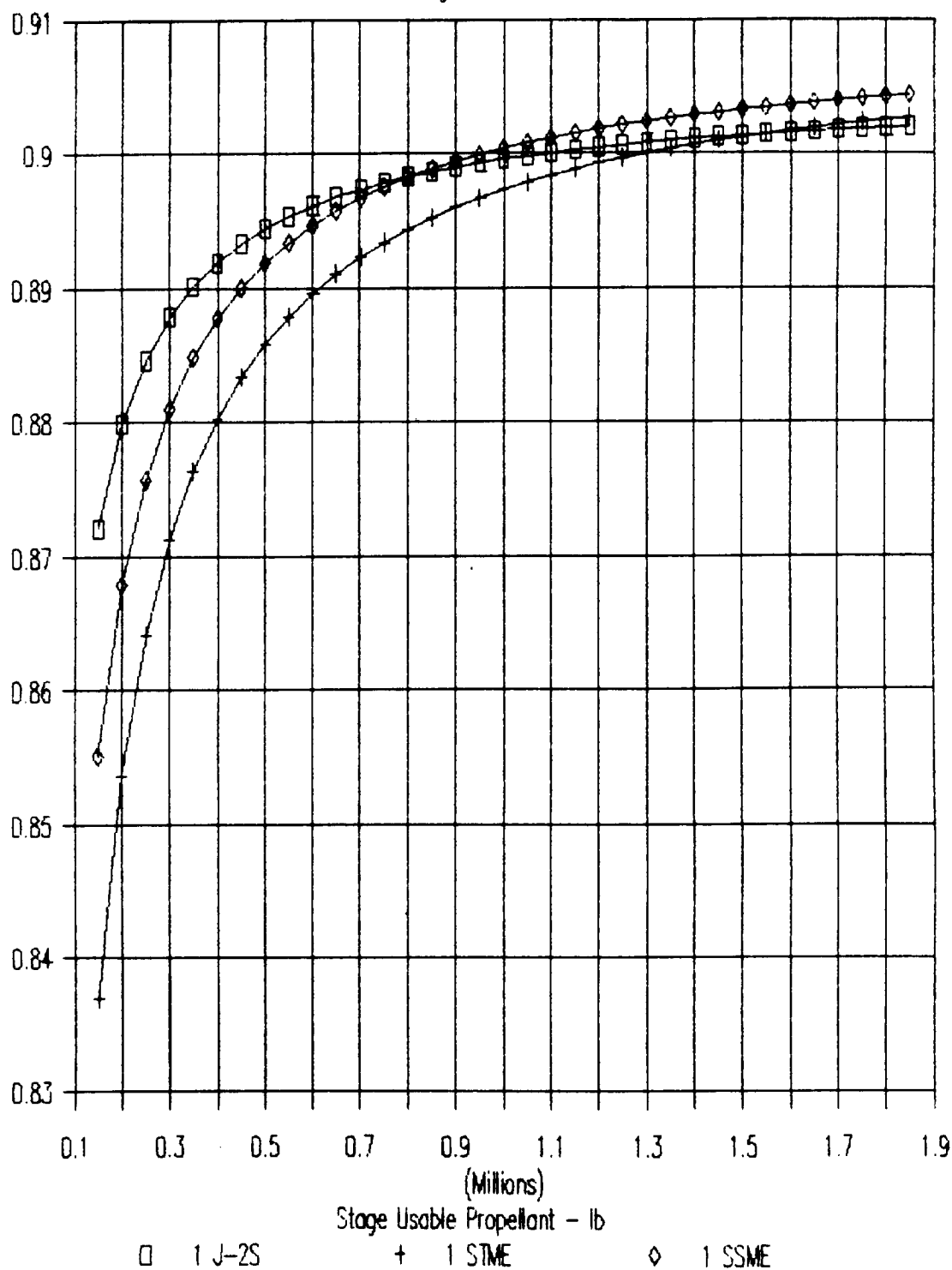
S-IVB DERIVED MASS FRACTIONS AS A FUNCTION OF STAGE PROPELLANT LOAD (Concluded)

One of the primary factors that affected the burn-out mass of the TLI stage was the engine selected. As an aid in the comparison of the effects on the engine chosen, propellant mass fraction curves for one J-2S engine, one SSME engine and one STME engine from the previous figures were replotted.

The results showed that a stage that used an SSME engine had a higher propellant mass fraction than a stage using a STME engine because a SSME and its associated thrust structure was lighter. A stage that used a J-2S engine started out as the lightest stage because of the lower engine and thrust structure mass. However, as additional propellant was added to the stages, the other stages became lighter because the lower propellant mixture ratio used by a J-2S engine caused the mass of the propellant tanks grow at a faster rate. The J-2S mixture ratio was 5.50 and the STME and SSME mixture ratios were both 6.0.

S-IVB Derived Upper Stage Performance

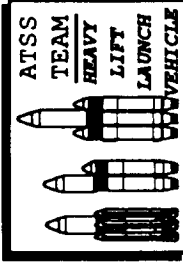
10% growth factor



FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

PROPULSION

Performance data for the F-1A and J-2S were obtained from a presentation on restarting the F-1 and J-2 production line, that was given to Lockheed by Terry Murphy, Manager of F-1 Program Development for Rocketdyne (818-718-4851). Additional information regarding F-1A maximum and minimum thrust levels, as demonstrated by actual engine test data, was obtained from Mr. Terry Murphy. Murphy stated that 100 percent RPL sea level thrust for the F-1A could be either 1,800,000 lbf or 2,000,000 lbf. Due to the range of booster engine numbers and propellant loads that Lockheed assessed, it was desired to have a range of F-1A thrust levels at 100 percent RPL to choose from, to better optimize first stage thrust-to-weight ratios for candidate configurations. While 75 percent RPL was the F-1A minimum throttle level typically quoted by the community, Murphy stated that 65 percent RPL was achievable for the F-1A at either of the two 100 percent thrust values. Murphy did not have a throttle-altitude constraint line for the F-1A, however. SSME data were obtained from brochures published by Rocketdyne. STME data were obtained from an NLS program summary briefing given by Porter Bridwell, MSFC's NLS Program Manager, dated 8 February 1992.



GROUND RULES AND CONSTRAINTS (Continued)

PROPULSION

STME Performance:

Max. Vac. Thrust = 650,000 lbf (100%RPL)
 Min. Throttle* = 70% RPL @ 13,500 ft. geodetic alt.
 Vac. lsp = 428.5 sec. (100% RPL)
 Min. lsp = 427.3 sec. (70 % RPL)
 Engine Mixture Ratio = 6.0:1
 Nozzle Expansion Ratio = 45.0:1
 Length = 160.0 in.
 Exit Diameter = 96.0 in.
 Weight = 9,930 lbf
 Pc = 2250 psia
 * (from 2/8/92 pitch by Bridwell/MSFC)

SSME Performance:

Max. Vac. Thrust = 470,000 lbf (100%RPL)
 Min. Throttle = 65% RPL
 Vac. lsp = 452.5 sec. (100% RPL)
 Engine Mixture Ratio = 6.0:1
 Nozzle Expansion Ratio = 77.5:1
 Length = 168.0 in.
 Exit Diameter = 96.0 in.
 Weight = 6,990 lbf
 Pc = 3000 psia

J-2S Performance*:

Max. Vac. Thrust = 265,000 lbf (100%RPL)
 Min. Throttle = 75% RPL (presumed level) @ vacuum
 Vac. lsp = 436.0 sec. (100% RPL)
 Engine Mixture Ratio = 5.5:1
 Nozzle Expansion Ratio = 40.0:1
 Length = 133.0 in. (from T. Murphy, Rocketdyne)
 Exit Diameter = 80.5 in. (from T. Murphy, Rocketdyne)
 Weight = 3,800 lbf (includes accessories)
 Pc = 12000 psia
 * (from 12/17/91 Rocketdyne pitch to LESC except where noted)

F-1A Performance*:

Max. Vac. Thrust = 2,020,500 lbf (100%RPL)
 Max. Sea Level Thrust = 1,800,000 lbf (100% RPL)
 Min. Throttle = 75% RPL @ sea level (presumed)
 Vac. lsp = 304.2 sec.
 Sea Level lsp = 271.0 sec.
 Engine Mixture Ratio = 2.27:1
 Nozzle Expansion Ratio = 16.0:1
 Length = 224.0 in.
 Exit Diameter = 143.5 in.
 Weight = 19,000 lbf
 Pc = 1161 psia
 * (from 12/17/91 Rocketdyne pitch to LESC)

FIRST LUNAR OUTPOST NLS-DERIVED LAUNCH VEHICLE ASSESSMENT

ENGINE LAYOUT

A common engine orientation was utilized for either 2-booster or 4-booster NLS-derived configurations. Engine gimbal clearance (avoiding bell-to-bell hard-over collisions), control authority, thrust structure complexity, and attach strut length penalties were the primary drivers for determining the layout of the booster and core engine clusters. Convective plume heating was acknowledged to be a secondary design consideration at the time of the analysis, and would require further assessment at a later time.

Placing the booster engines on the propellant tank perimeter simplified the thrust structure required to shear the thrust loads into the aft propellant tank (RP-1 tank), but restricted how close the boosters could be clustered about the core vehicle. While being highly desirable to accept the STME positioning as currently baselined by the NLS program for the NLS-1 vehicle, it was found that the best compromise between booster and core engine locations and attach strut length was to locate the STMEs slightly inboard of the core tank perimeter. A minimum dynamic clearance of two inches was maintained between either booster or core engines, to allow for thrust vector control actuator dither. The desired locations of the engines on the boosters and core were closely coupled with the design of the thrust vector control subsystem, and became part of an iterative solution when considering attach strut lengths and plume heating. If convective plume heating between F-1A pairs (upper or lower engine pairs, in a local vertical/local horizontal sense) requires the F-1As to be spaced farther apart, the boosters will not be able to be placed as close to the core vehicle, necessitating longer attach struts and more core vehicle skin stiffening at the attach struts.

The booster engines were spaced relative to each other to allow for all four F-1As to be gimballed if the booster was to be used as a stand-alone launch vehicle. The resulting gimbal traces of either the were then designed to be within two inches of each other. It was also assumed that aerodynamic fairings would be required for each booster engine, since their locations on the booster perimeter would place the engine bells into the freestream flow during first stage ascent. Since the core engines were not required to gimbal for thrust vector control during first stage, their inboard location on the core vehicle would shield them from the first stage freestream flow, therefore not requiring the use of aerodynamic fairings. Removal of core engine fairings allowed the boosters to be placed closer to the core.

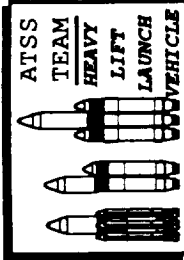
FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT
THRUST VECTOR CONTROL

As was mentioned in the discussion of the booster and core vehicle engine placement, the method of performing thrust vector control (TVC) was closely coupled with engine placement, attach strut, and plume heating design issues. It was presumed that for both the two and four-booster configurations, the boosters would provide directional steering through the gimballing of booster engines during first stage, given their dominant control authority. The use of either hydraulics or electromechanical actuators for TVC was not a first-order design driver on the engine placement and gimbal frame (rock/tilt or pitch/yaw).

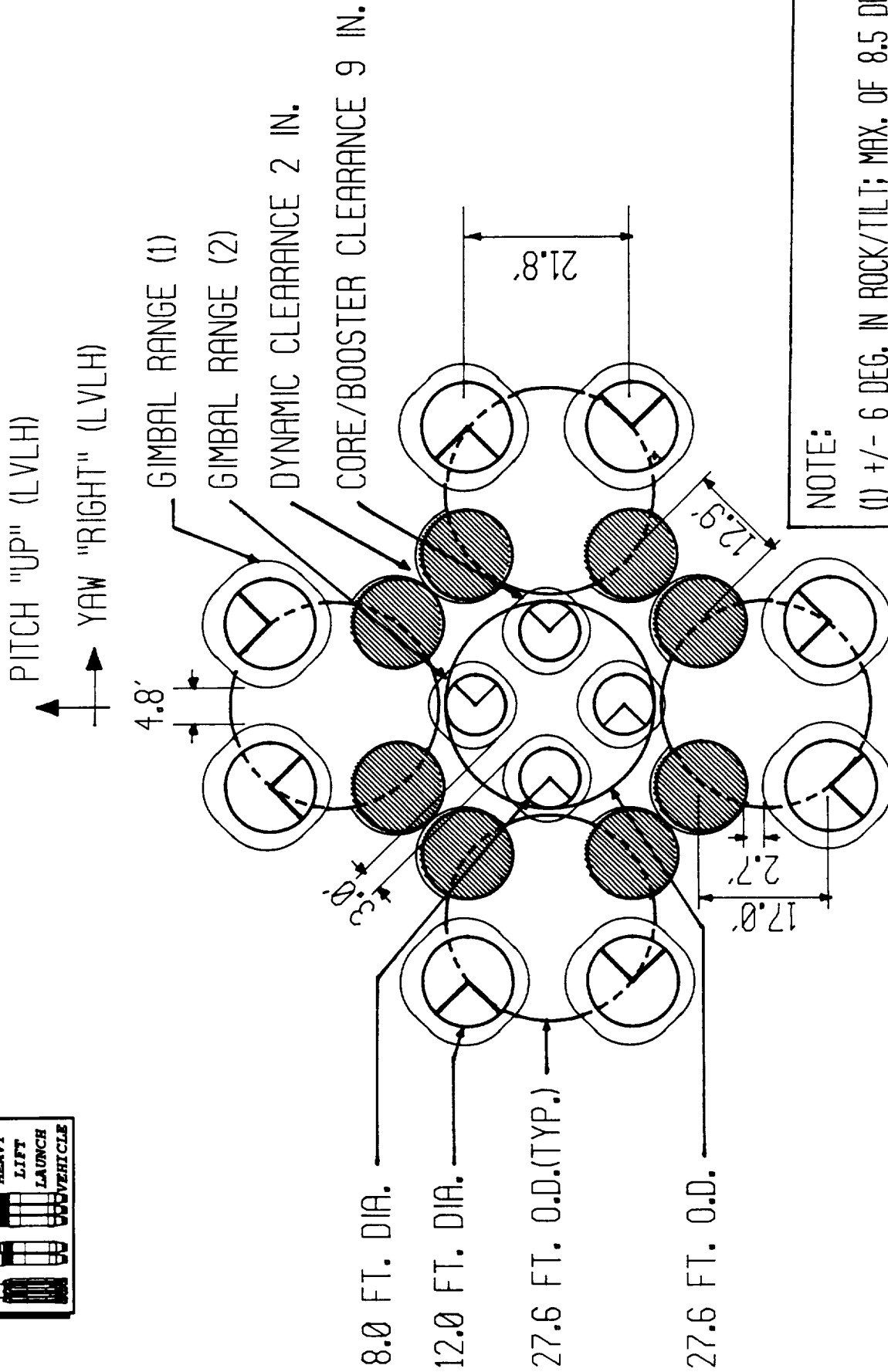
A rock/tilt gimbal orientation (relative to local vertical/local horizontal) was chosen for both the boosters and core vehicle over a pitch/yaw orientation. By placing the gimbal patterns in a rock/tilt orientation, effective gimbal throw lengths were maximized in the two dominant ascent control planes (pitch and yaw). Such an arrangement was also presumed to allow sufficient control authority in all three control axes if one outboard booster engine was to be shut down on opposing boosters.

The core vehicle engines contained rock/tilt gimbal actuation, but were not required to gimbal during first stage. During booster separation, classically the core vehicle engines are commanded to an attitude hold until guidance has a converged solution and the boosters have fallen to a safe distance behind the core vehicle. Freezing the core gimbals during first stage helped the desire to place the boosters as close to the core vehicle as possible. The next major engine bell-to-bell collision avoidance issue became between engines on opposing boosters. To again facilitate minimum attach strut lengths, the "interior" booster engines were designed for no gimballing. It was felt that there was more than adequate control authority in pitch, yaw, and roll by only gimballing the outboard booster engines.

The gimbal angle requirement baselined by NLS (+/- 6 degrees in rock/tilt, with a maximum effective angle of 8.5 degrees in pitch or yaw) was adopted for both the booster and core vehicle gimbal ranges. The resulting gimbal traces were identified to insure that gimbal hard-overs on adjacent engines would not cause a bell-to-bell collision. A minimum two inch dynamic clearance was always maintained between adjacent engines or their respective gimbal traces. The gimbal orientations for the boosters and core were designed to be the same for either the two or four-booster configurations. Six degree-of-freedom control authority simulations were not exercised as part of the design activity due to study time limitations. Such simulations will be required during more in-depth trade study assessments of the launch vehicle design.



EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

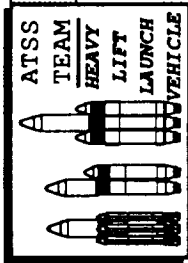


- SHADED BOOSTER ENGINES DO NOT GIMBAL
- CORE ENGINES DO NOT GIMBAL IN FIRST STAGE

FIRST LUNAR OUTPOST
NLS-DERIVED LAUNCH VEHICLE ASSESSMENT
SUMMARY OF CANDIDATE HLLV CONFIGURATIONS

Over thirty different HLLV configurations were sized and their performance assessed. When striving for the design goal of 93 metric tons of payload post-TLI, adhering to current KSC facility physical constraints (principally the VAB high bay door vertical clearance limit), and using ET-based booster and core diameters, there were three two-stage configurations (four four-engined boosters) that met the requirements. Their payload capabilities post-TLI, depending upon core propellant load (5 foot or 15 foot LH2 tank extension) and TLI propellant load (590,000 lbm or 760,000 lbm), were 106-120 metric tons. A three-stage core with two boosters met the payload goal but caused the VAB high bay door clearance limit to be exceeded. A Mars mission configuration was also identified, in which the TLI stage was replaced by a kick stage (two Shuttle Orbital Maneuvering System engines) and flown to a 220 nautical mile due-east orbit. When adhering to the VAB door constraint, the Mars configuration could not meet the desired 200-250 metric ton payload goal; achieving only 186 metric tons.

The assessment of the candidate configurations, while thorough, still allowed room for further performance enhancements, such that the performance capabilities contained herein only represent a conservative first estimate.



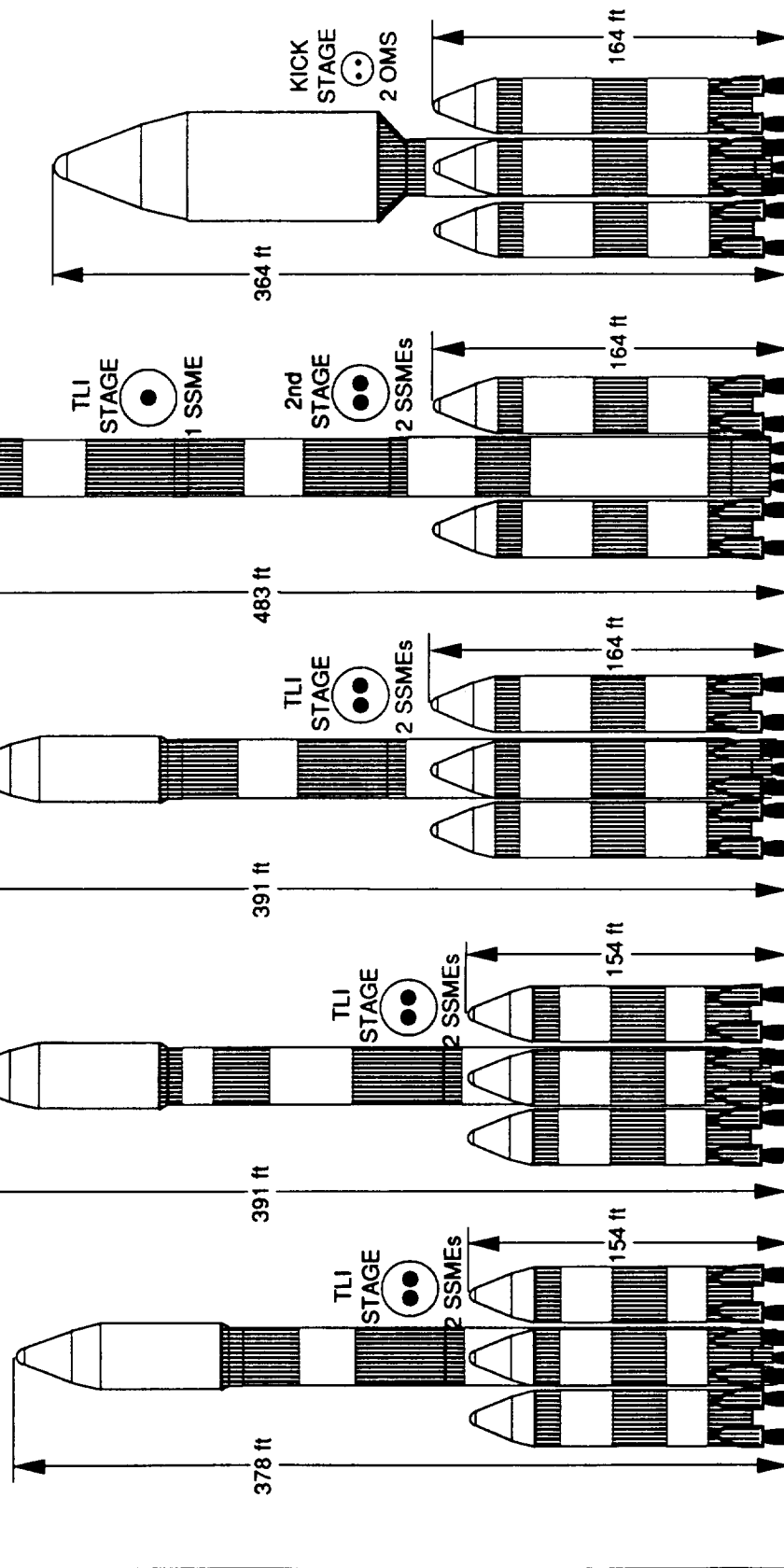
EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

SUMMARY OF CANDIDATE HLLV CONFIGURATIONS

For all configurations:
 Core: 4 STMEs (650K lbf vac., 100% RPL)
 Booster: 4 F-1As (1.8M lbf s.l., 100% RPL)
 For upper stages:
 SSME 470K lbf vac. (100% RPL)

VAB HIGH BAY DOOR VERTICAL CLEARANCE ENVELOPE (APPROX.)

390-400 ft



ET core +5 ft
 590K TLJ prop.

PAYLOAD POST-TLJ 106 mt

ET core +5 ft
 760K TLJ Prop.

109 mt

ET core +15ft
 590K TLJ prop.

120 mt

ET core +15 ft
 590K 2nd Sig. prop.
 590K TLJ prop.

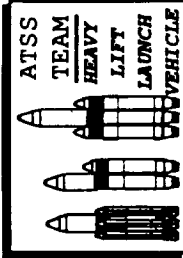
107 mt

ET core + 15 ft
 20K Kick stg. prop.

186 mt

* To LEO

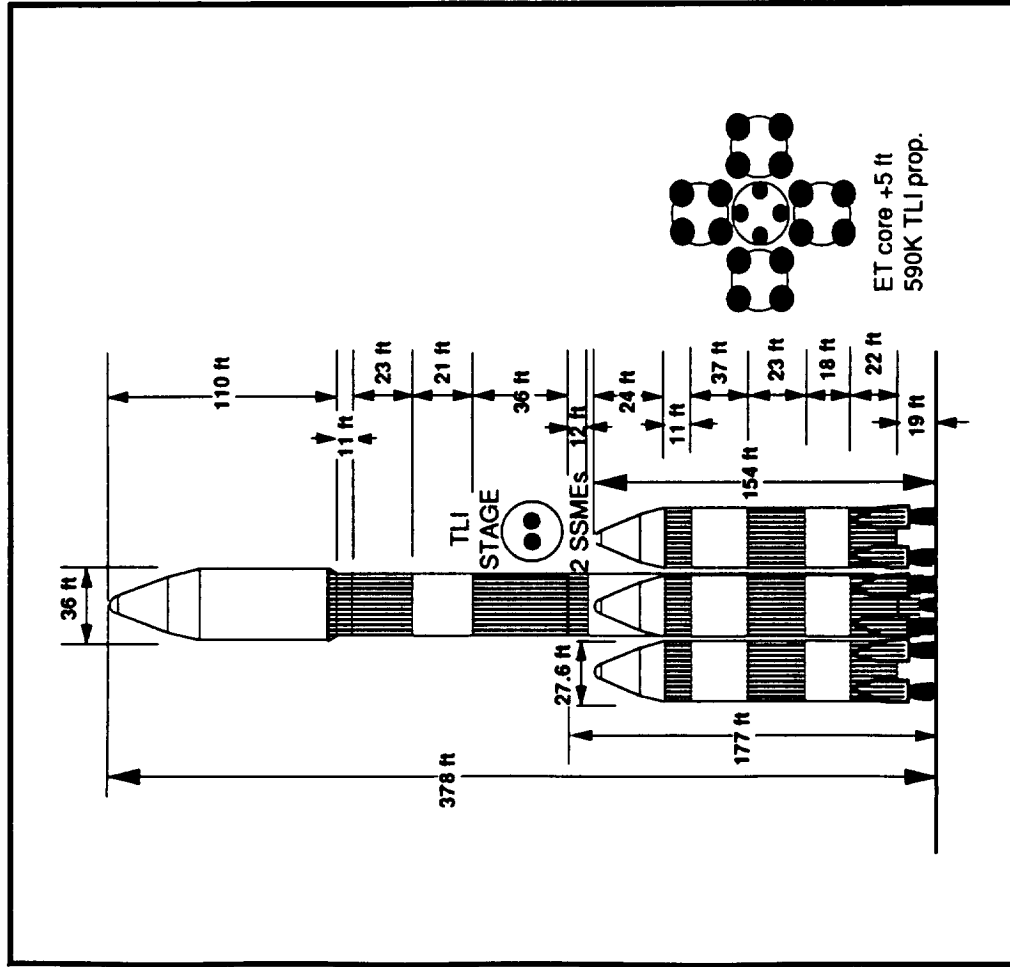
McCurry
 3/19/92



EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

Lunar Vehicle

Standard NLS Core w/ LOX/RP1 Boosters
and TLI stage (590 k lbm propellant)



Payload:	235 K lbm (106 t)
Final Position:	TLI

GLOW: 15,359,000 lbm

CORE:

Inert Mass: 187.8 K lbm
Propellant Mass: 1.69 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 173 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusability: N.A.

BOOSTER:

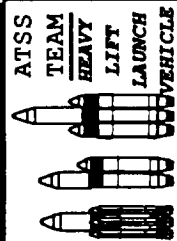
Number/Type: 4/ET+ 5 ft
Inert Mass: 235 K lbm
Propellant Mass: 2.90 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 154 ft
Diameter: 27.6 ft
Reusability: N.A.

TLI Stage:

Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusability: N.A.

SHROUD – Usable Volume: 33 x60 ft; Mass: 35,500 lb
Comments: • ET prop. capacity based on a 5 ft stretch
• STME has 75% step throttle
• Max. G = 4.5 Max. Q = 900 psf

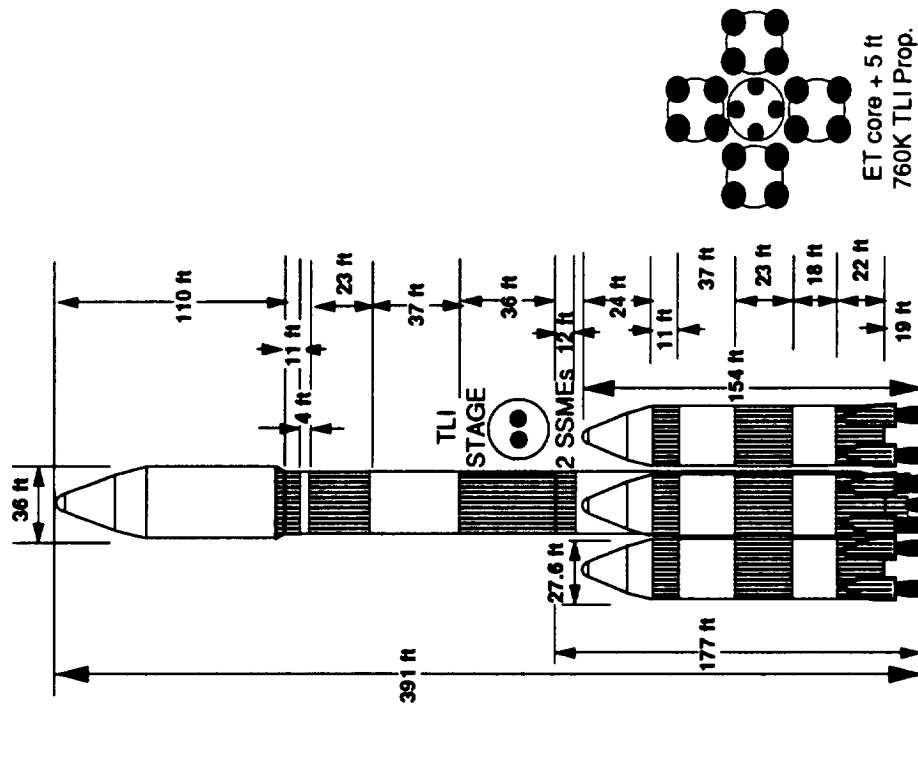




EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

Lunar Vehicle

**Standard NLS Core w/ LOX/RP1 Boosters
and TLI stage (760 k lbm propellant)**



SHROUD – Usable Volume: 33 x60 ft; Mass: 35,500 lb

Comments:

- ET prop. capacity based on a 5 ft stretch
- STME has 75% step throttle
- Max. G = 4.5 Max. Q = 900 psf



McCurry
3/19/92

Payload: Final Position:	240 K lbm (109 t) TLI
---	--

GLOW: 15,551,000 lbm

CORE:

Inert Mass: 187.8 K lbm
Propellant Mass: 1.69 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 173 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

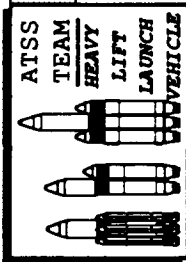
Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusability: N.A.

BOOSTER:

Number/Type: 4/ET+ 5 ft
Inert Mass: 235 K lbm
Propellant Mass: 2.90 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 154 ft
Diameter: 27.6 ft
Reusability: N.A.

TLI Stage:

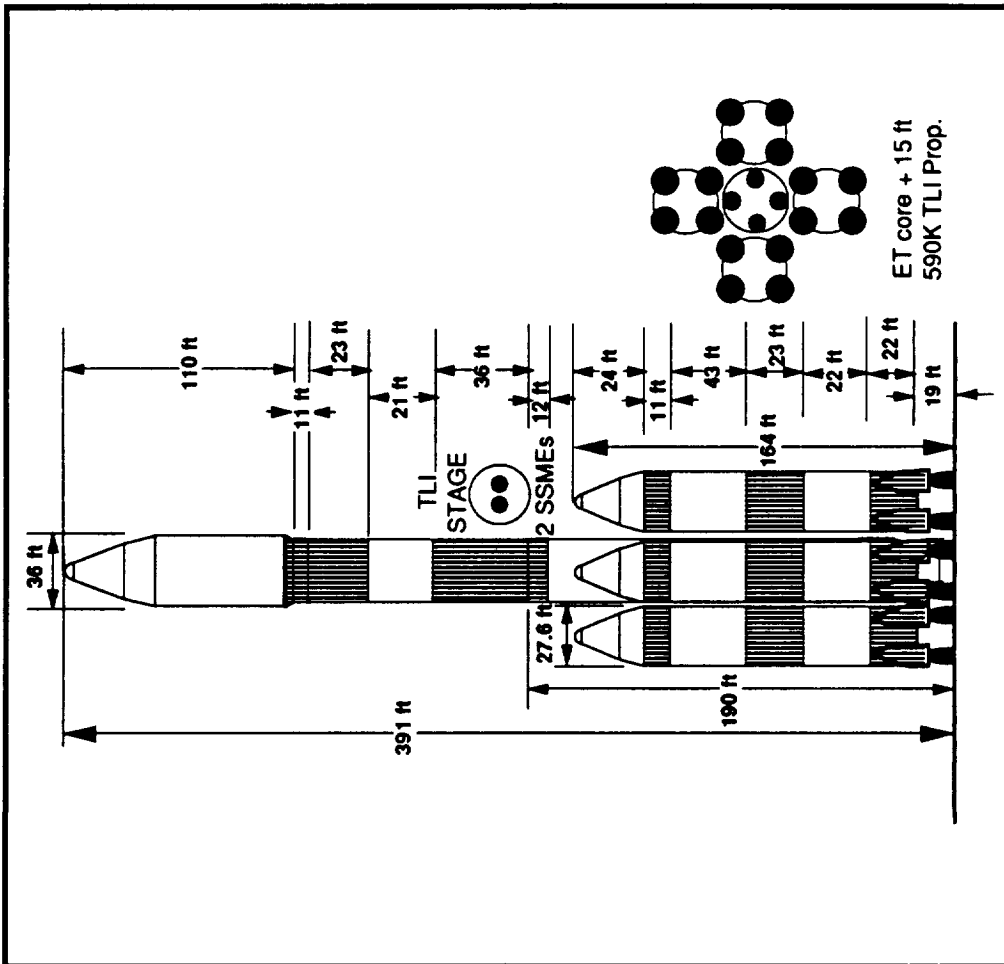
Inert Mass: 95.9 K lbm
Propellant Mass: 0.76 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5sec
Engine Exit Dia.: 96 in.
Length: 101 ft
Diameter: 27.6 ft
Reusability: N.A.



EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

Lunar Vehicle

Stretched NLS Core w/ LOX/RP1 Boosters
and TLI stage (590 k lbm propellant)



SHROUD – Usable Volume: 33 x60 ft; Mass: 35,500 lb
Comments: • ET prop. capacity based on a 15 ft stretch
• STME has 75% step throttle
• Max. G = 4.5 Max. Q = 900 psf



Payload: 265 K lbm (120 t)
Final Position: TLI

GLOW: 17,077,000 lbm

CORE:

Inert Mass: 206.4 K lbm
Propellant Mass: 1.86 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 186 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusability: N.A.

BOOSTER:

Number/Type: 4/ET + 15 ft
Inert Mass: 251 K lbm
Propellant Mass: 3.26 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 164 ft
Diameter: 27.6 ft
Reusability: N.A.

TLI Stage:

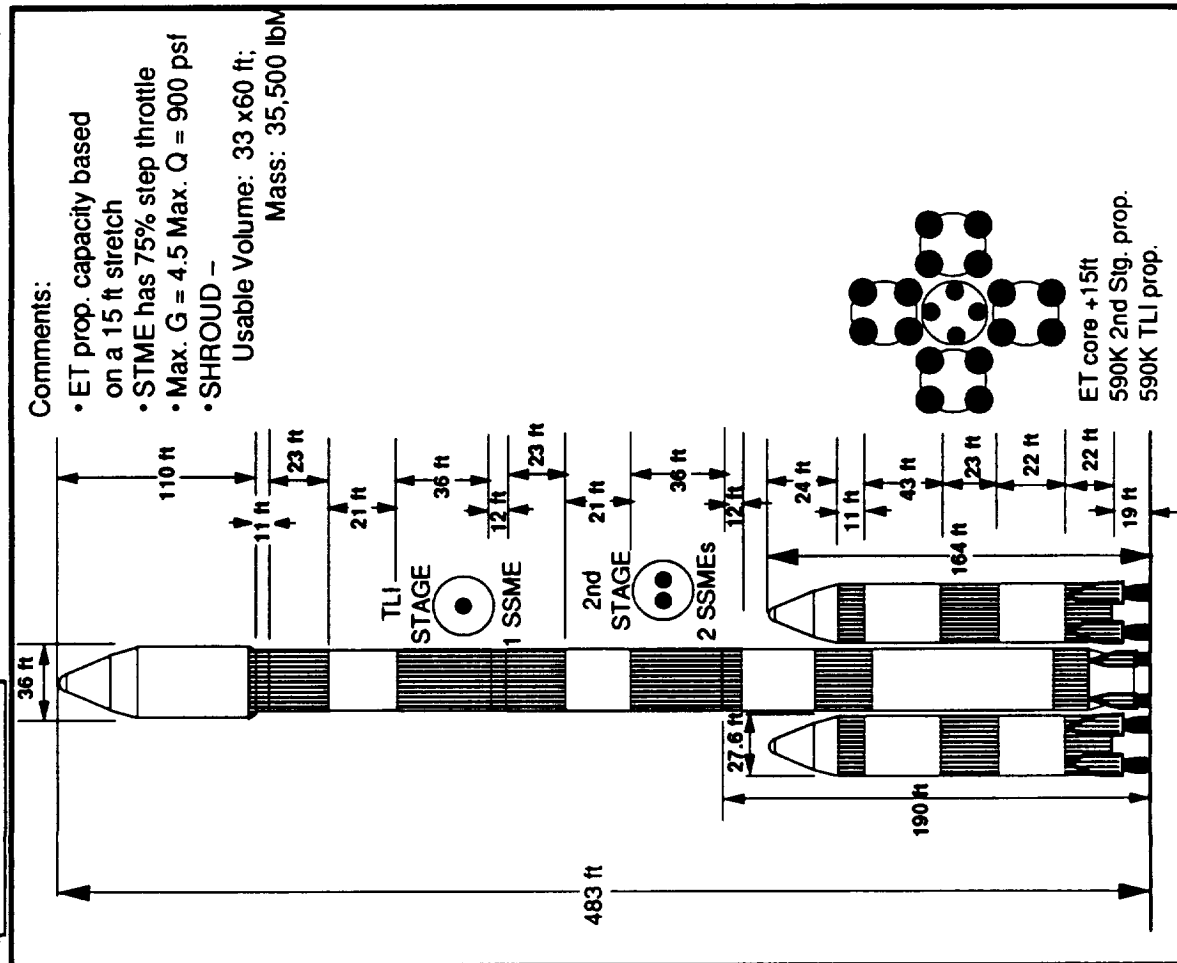
Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusability: N.A.

ATSS TEAM HEAVY LIFT LAUNCH VEHICLE

EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

Lunar Vehicle

Stretched NLS Core w/ LOX/RP1 Boosters, 2nd Stage,
(590 k lbm propellant) and TLI stage (590 klbm
propellant)



Payload:	236 K lbm (107 t)
Final Position:	TLI

GLOW: 10,686,000 lbm

CORE:

1st Stage:
Inert Mass: 206.4 K lbm
Propellant Mass: 1.86 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 186 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

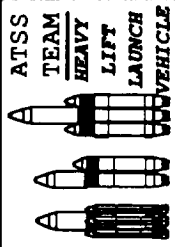
Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusability: N.A.

BOOSTER:

Number/Type: 2/ET+ 15 ft
Inert Mass: 251 K lbm
Propellant Mass: 3.26 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 164 ft
Diameter: 27.6 ft
Reusability: N.A.

TLI Stage:

Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/1
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusability: N.A.



EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

Mars Vehicle

Stretched NLS Core w/ LOX/RP1 Boosters
and Kick Stage (14.3 K lbm propellant)

Payload:	410 K lbm (186 t)
Final Position:	220 nm LEO

GLOW: 16,696,000 lbm

CORE:

Inert Mass: 206.4 K lbm
Propellant Mass: 1.86 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 186 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

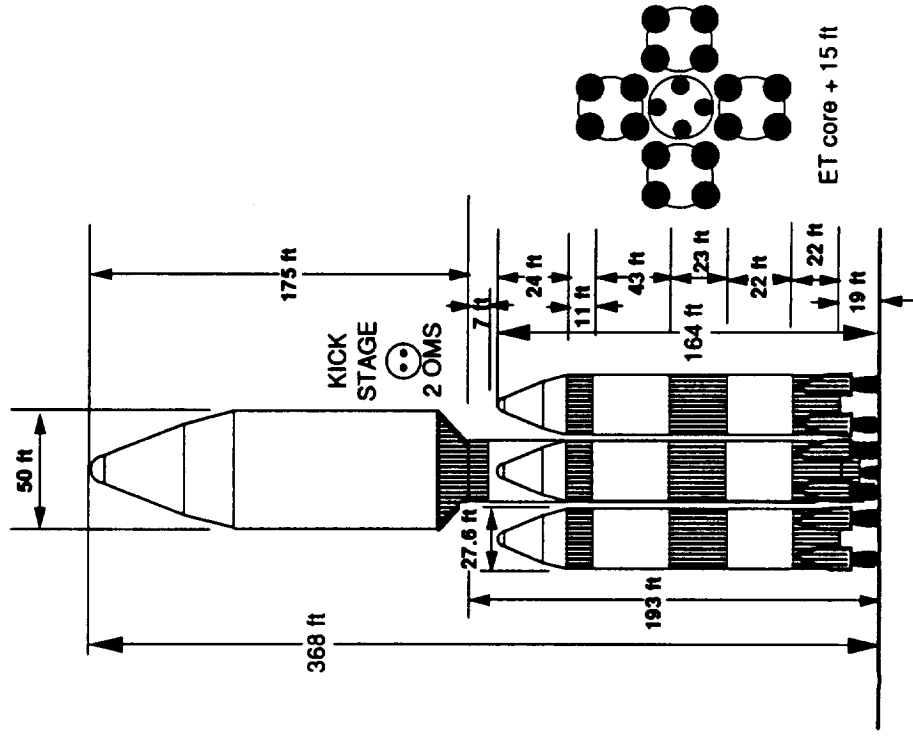
Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusability: N.A.

BOOSTER:

Number/Type: 4/ET + 15 ft
Inert Mass: 251 K lbm
Propellant Mass: 3.26 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 164 ft
Diameter: 27.6 ft
Reusability: N.A.

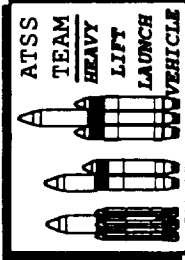
KICK STAGE:

Inert Mass: 3.6 K lbm
Propellant Mass: 14.3 K lbm
Propellant Type: NTO/MMH
Engine Type/No.: OMS/2
Vac Thrust (ea.): 6.0 K lbf
Vac ISP: 313.0 sec
Engine Exit Dia.: 21.5 in.
Length: 6 ft
Diameter: 27.6 ft
Reusability: N.A.



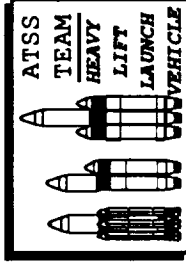
SHROUD - Usable Volume: 50 x 100 ft, Mass: 160,000 lb
Comments: • ET prop. capacity based on a 15 ft stretch
• STME has 75% step throttle
• Max. G = 4.5 Max. Q = 900 psf





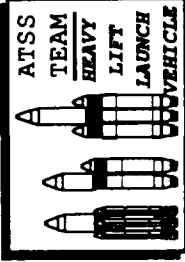
EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

CONCLUSIONS



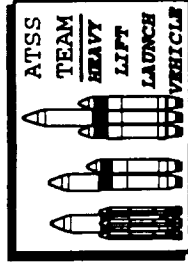
CONCLUSIONS

- To meet a lunar mission requirement of >90 mt post-TLI and not violate the VAB high bay door clearance, must use four F-1A powered boosters with the NLS-derived core
- Use of four F-1As on four NLS-derived boosters with 2 SSMEs on the TLI stage provide approximately 115 mt of payload post-TLI
- Use of F-1A throttling for Qbar control gains approximately 30,000 lbm of payload post-TLI
- Use of STMEs for upper stage applications cannot compete with SSMEs from vehicle sizing and performance standpoints
- Use of 2 boosters with 3 F-1As (1.8E6 lbf s.l. thrust) will not allow greater than 90 mt post-TLI, without using both a second stage and a TLI stage
- Use of common propellant tank, intertank, aft skirt, and forward skirt/interstage piece-parts still allows reasonable vehicle configurations to be designed, without incurring inordinate performance losses
 - Associated cost savings could offset non-optimality of the design
- The VAB high bay doors can be economically modified, up to a point, to accommodate booster height, but not economically for core vehicle height



EARLY RETURN TO MOON/MARS: NLS-DERIVED HLLV OPTIONS

AUXILIARY CONCLUSIONS

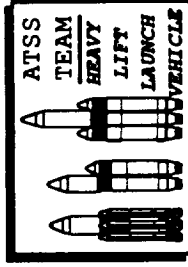


CONCLUSIONS

Performance

- To meet a lunar mission requirement of >90 mt post-TLI and not violate the VAB high bay door clearance, must use four F-1A powered boosters with the NLS-derived core
- Use of four F-1As on four NLS-derived boosters with 2 STMEs on the TLI stage provide approximately 100 mt of payload post-TLI
- Use of four F-1As on two NLS-derived boosters with 2 STMEs on the TLI stage provide approximately 63 mt of payload post-TLI
- Use of four F-1As on two NLS-derived boosters with 2 SSMEs on the TLI stage provide approximately 77 mt of payload post-TLI
- Use of four F-1As on four NLS-derived boosters with 2 SSMEs on the TLI stage provide approximately 120 mt of payload post-TLI (with 15 ft. tank extension)
- Use of F-1A throttling for Qbar control gains approximately 30,000 lbm of payload post-TLI
- Use of STMEs for upper stage applications cannot compete with SSMEs from vehicle sizing and performance standpoints

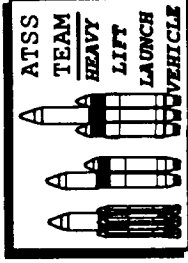




CONCLUSIONS (Continued)

Performance (Concluded)

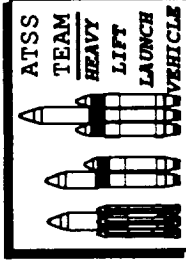
- Use of 2 boosters with 3 F-1As (1.8E6 lbf sea level thrust) will not allow greater than 90 mt post-TLI, without using both a second stage and a TLI stage
- Use of 4 F-1As and no second stage requires a core LH2 tank stretch of 10-15 feet over the current ET, in order to better utilize the extra booster thrust (>15 foot stretch is probably performance optimal)
- Liftoff thrust/weight ratio requirements ranged from 1.4 to 1.6
- Booster separation thrust/weight ratio requirements had a minimum of 0.8
- Second stage ignition thrust/weight ratio requirements had a minimum of 0.7
- Suborbital TLI stage ignition thrust/weight ratio requirements had a minimum of 0.6
- Minimizing vehicle dry weight (and hence cost) tends to move the velocity split from the TLI stage to the second stage (if such a stage is used)
- Use of Aluminum-Lithium (8090 or Ultralite) would provide 10-30 % weight savings over the majority of the launch vehicle



CONCLUSIONS (Continued)

Flight Mechanics

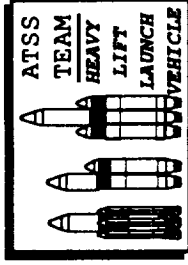
- Dynamic pressure (Q_{bar}) limiting is required for structural and thermal considerations
 - Lofting: more direct control but incurs significantly higher performance losses (gravity loss, thrust vector loss)
 - Throttling: less direct control but has significantly lower velocity losses
- Thrust acceleration (G) limiting is required for structural considerations
 - Throttling: effective control for minimum throttles of 65-75 %
 - Engine Shut-down: less precise control, and requires multiple shut-downs for moment balance and thrust vector loss minimization
- For the configurations assessed, only small exceedances over 4 Gs during ascent were observed with engine shut-down



CONCLUSIONS (Continued)

Manufacturability

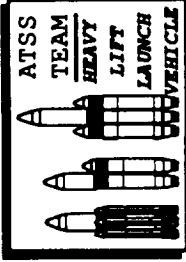
- Use of common propellant tank, intertank, aft skirt, and forward skirt/interstage piece-parts still allows reasonable vehicle configurations to be designed, without incurring inordinate performance losses
 - Associated cost savings should justify non-optimality of the design
- If the NLS-derived core has its propulsion module integrated at MAF, then the following tooling/facilities cost impacts are predicted by MMMSS (Bill Clayborn) for LH2 tank stretches:
 - 10 foot stretch over ET: approximately \$65 million non-recurring
 - 15 foot stretch over ET: approximately \$120 million non-recurring
- Tooling/certification requirements and LOX compatibility issues remain to be answered for use of Aluminum-Lithium
 - Lockheed makes the Titan IV conical adapter out of 8090 currently
- Common bulkhead manufacturing issues remain to be answered



CONCLUSIONS (Continued)

Manufacturability (Concluded)

- SSME pre-launch thermal conditioning is required for either a second stage or a TLI stage
- No SSME thermal conditioning is required during ascent for a suborbital burn of either a second stage or TLI stage
- SSME thermal conditioning is required for an on-orbit burn of a TLI stage (presuming 1-3 rev.s on-orbit prior to TLI ignition)
- SSME on-orbit thermal conditioning requirements (from Erv Eberly/Rocketdyne)
 - Approximately 100 lbm for recirculation pumps (1 per 2 SSMEs)
 - Approximately 1000 lbm each of LOX & LH2 per engine for thermal conditioning
- Air-start thermal conditioning requirements for STME remains to be answered
 - Dick McMillian/Rocketdyne contacted per STME

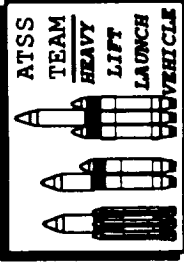


CONCLUSIONS (Continued)

Operations

- If build a new VAB for lunar/Mars applications, the core vehicle should be stretched to its maximum allowable length (cost constrained), and use SSMEs on a second stage and a TLI stage for payload maximization
- The combination of the VAB high bay door vertical clearance constraint (390-400 feet range) and the final mission payload requirements may require the use of common bulkheads for propellant tank construction
- Widening of the VAB high bay door opening (beyond 71 feet) will be more cost effective than rotating the orientation of a four-booster stack on the MLP in order to roll into the VAB
- --A rotation will work for VAB clearance but will incur significantly higher design/cost impacts at the launch pad
- --A 45 degree rotated orientation would be feasible from a T-0 umbilical and an ascent performance standpoint
- Utilizing one type of engine for the core vehicle would be preferred over multiple types of engines
- Pre-launch thermal conditioning of any air-startable engines can be accomplished but complicates ground processing

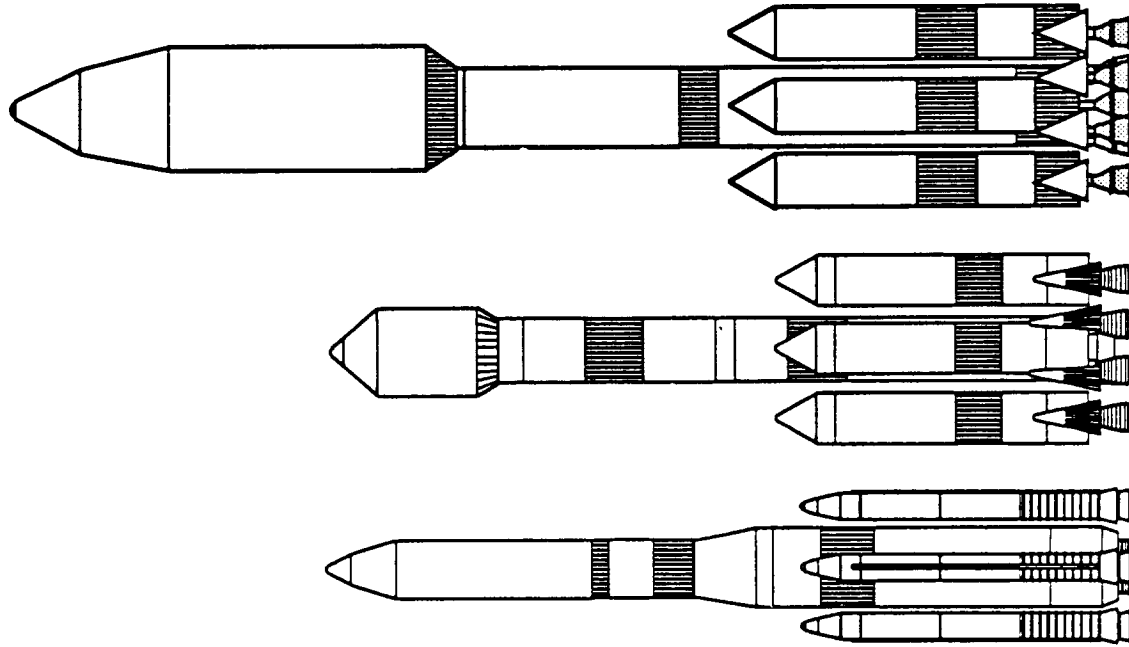




CONCLUSIONS (Concluded)

Cost

- Cost benefit tradeoffs remain to be performed on marginal cost of DDT&E cost reduction versus recurring cost reduction
- The cost of providing throttle-down capability on the boosters should be largely offset by the resulting increase in payload capability (as costed in terms of equivalent numbers of flights)
- The VAB high bay doors can be economically modified, up to a point, to accommodate booster height, but not economically for core vehicle height
- Approximate cost for a lunar/Mars VAB of \$375 million (FY92 \$s)

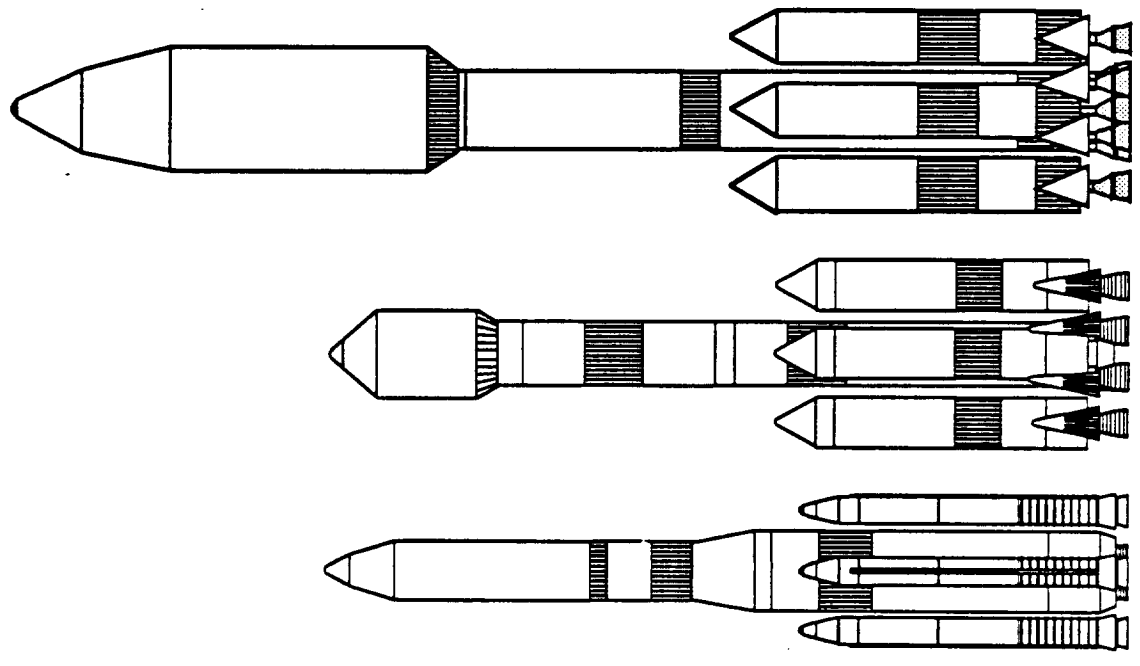
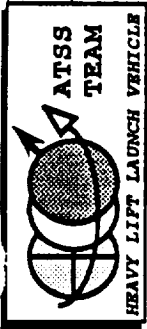


TECHNICAL AREA 2 HEAVY LIFT LAUNCH VEHICLE DEVELOPMENT

Concurrent Engineering
Meeting #1
5-6 May 1992

LMSC-MSD
Sunnyvale, CA





INTRODUCTION

--Introduction of Attendees

--FLO Organization





Exploration Programs Office — Organization —



Office of Exploration

NASA Headquarters
Office of Exploration
Michael Griffin, AA
Jay Greene, Deputy

Johnson Space Center
Aaron Cohen, Director
P.J. Weitz, Deputy

Manager
Douglas R. Cooke
Deputy Manager
Humboldt C. Mandell, Jr.
Deputy for Science
Michael B. Duke

**Mission Analysis and
System Engineering**
Dwayne Weary, Manager

Program Integration
Phil Deans, Manager

**Exploration Program
Development and Control**
Richard Fox, Manager

Major Activities:

Lunar Scout
Mike Conley

Artemis
Dave Saucier

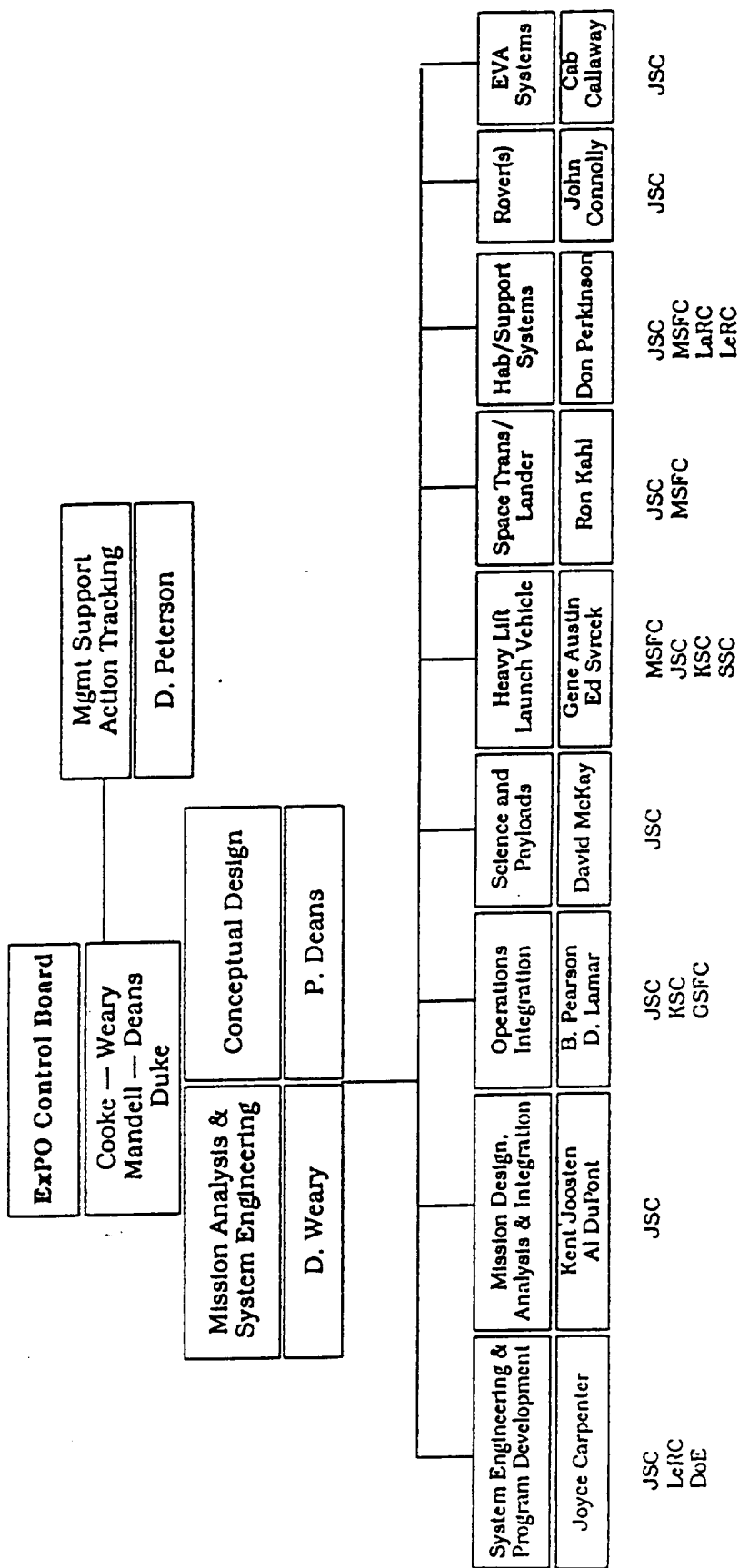
**First Lunar
Outpost**
Weary/Deans

Mars Exploration
Mike Duke



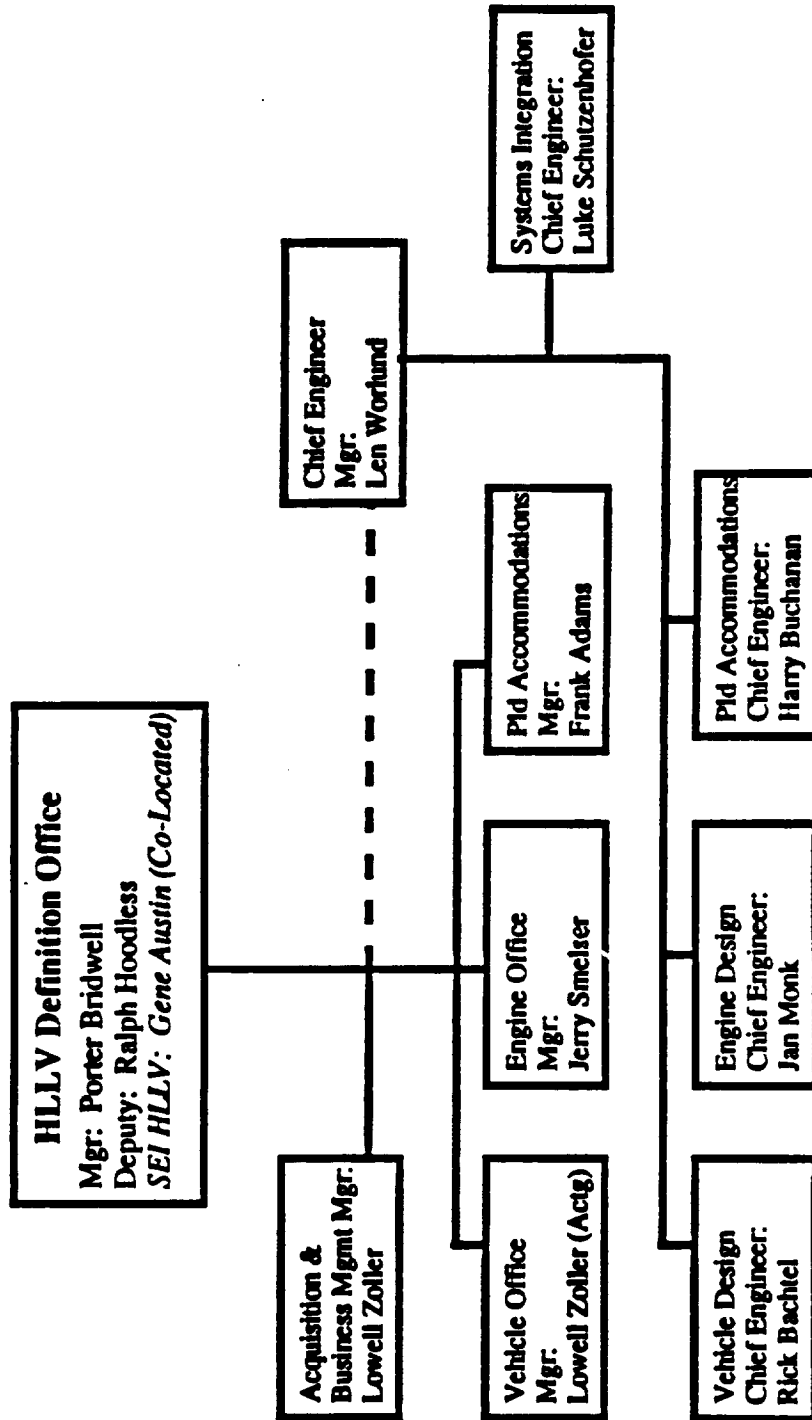
First Lunar Outpost Study Team

Office of Exploration



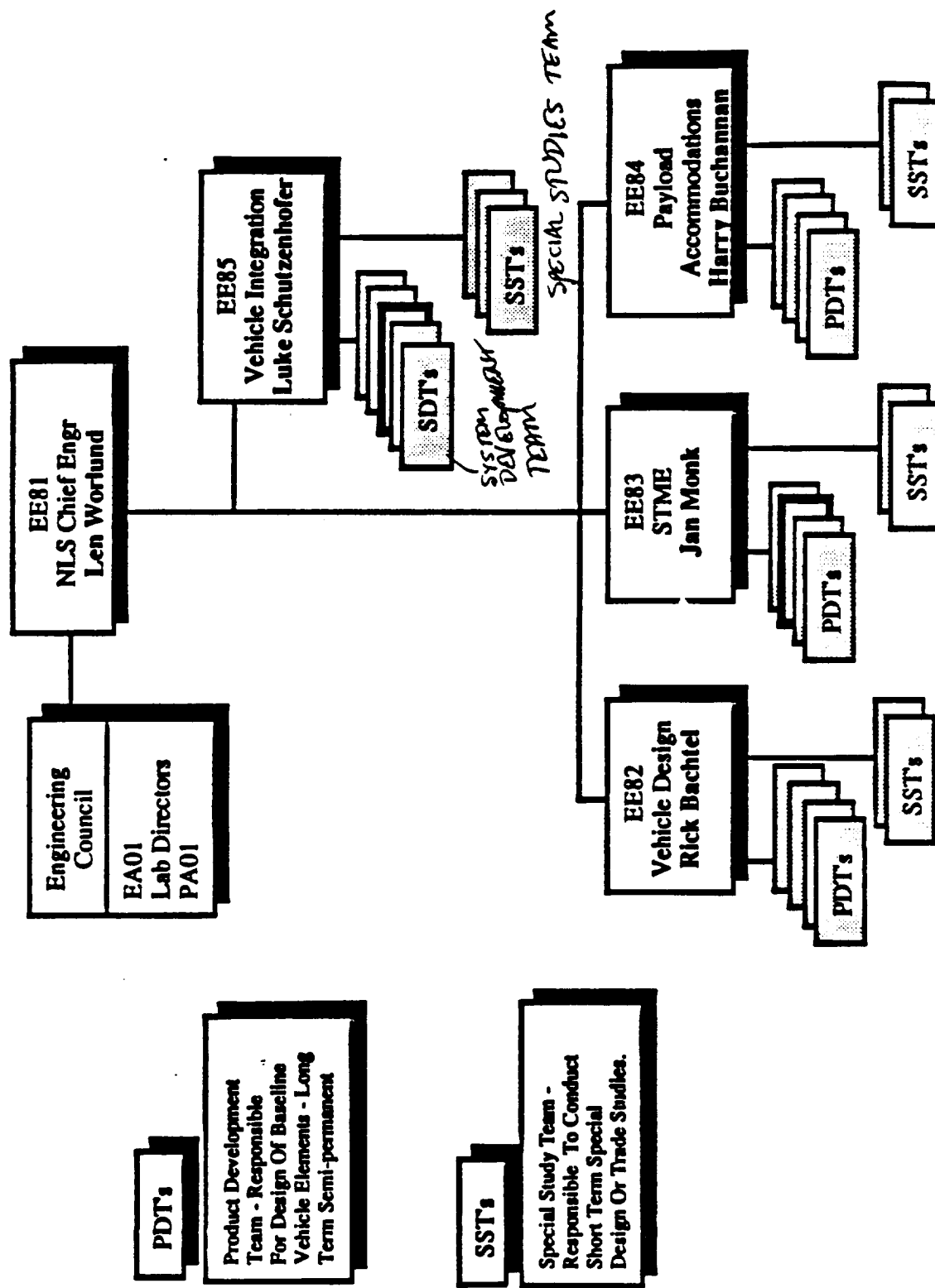


HLLV Definition Office



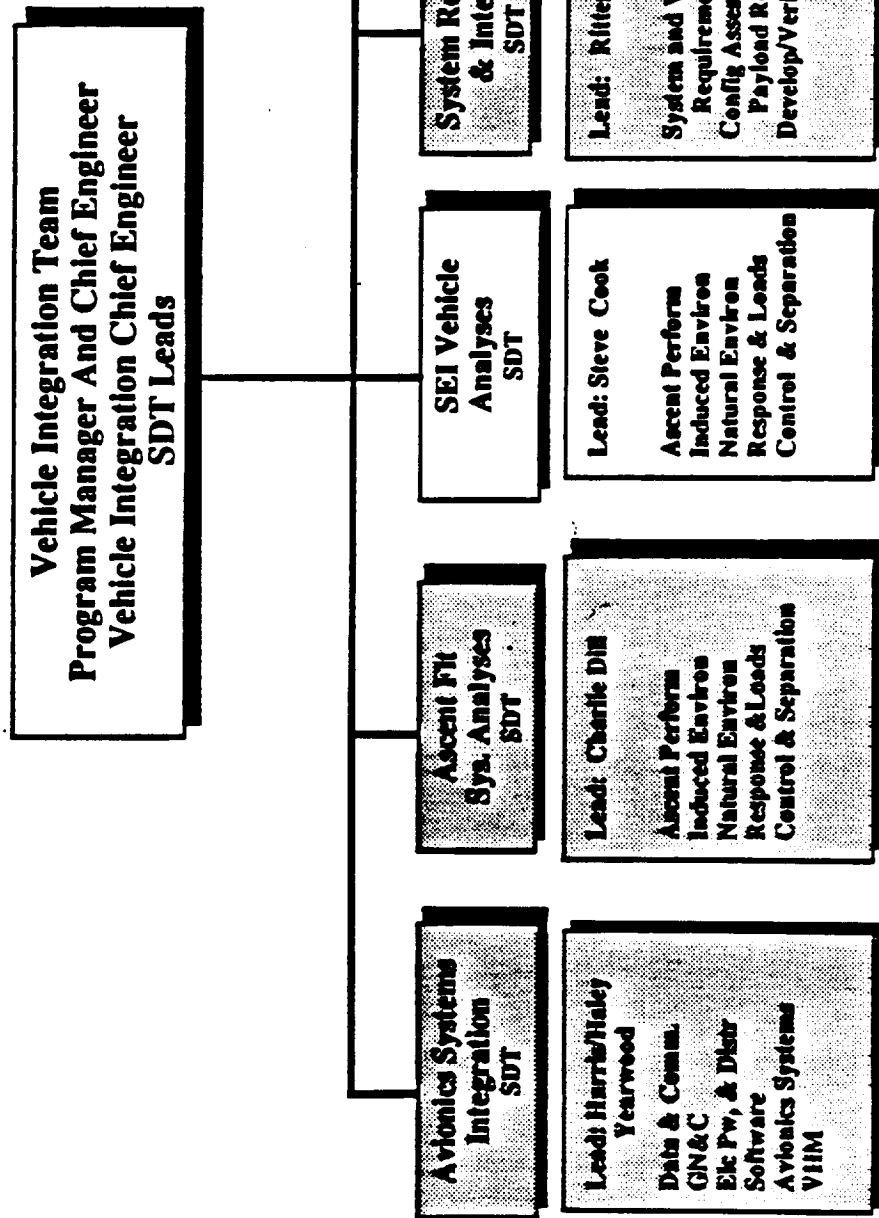


NLS S&E Organization





Vehicle Integration Team

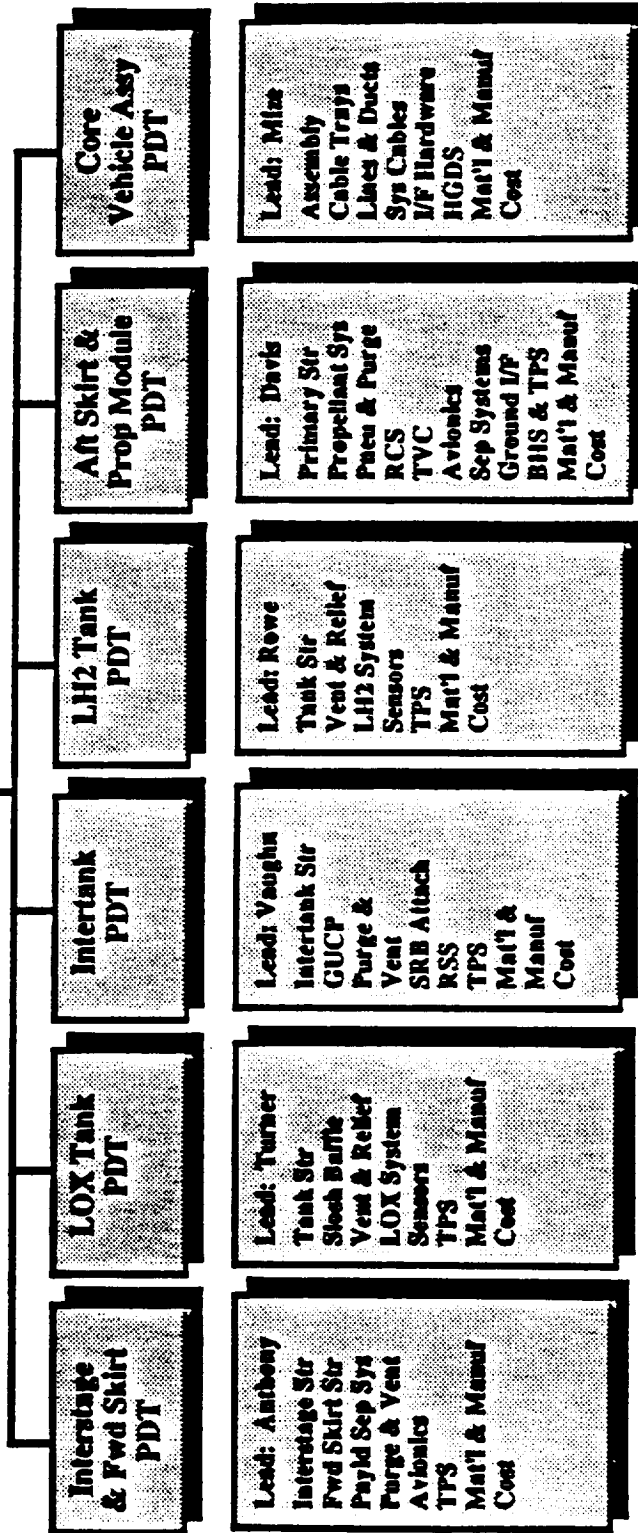




Core Vehicle Design Team

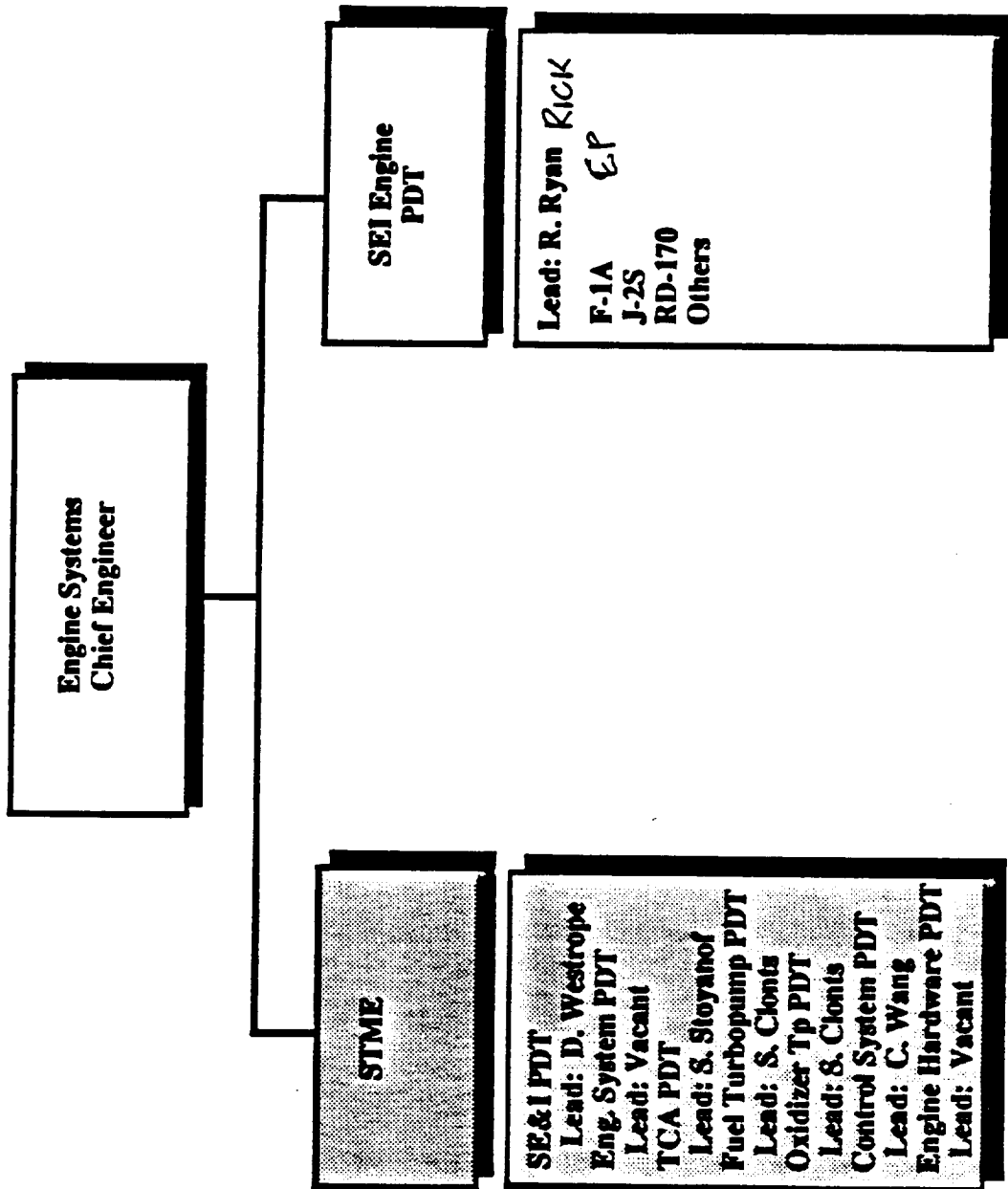


Core Vehicle Design Team
Vehicle Project Manager
Vehicle Chief Engineer
PDT Leads



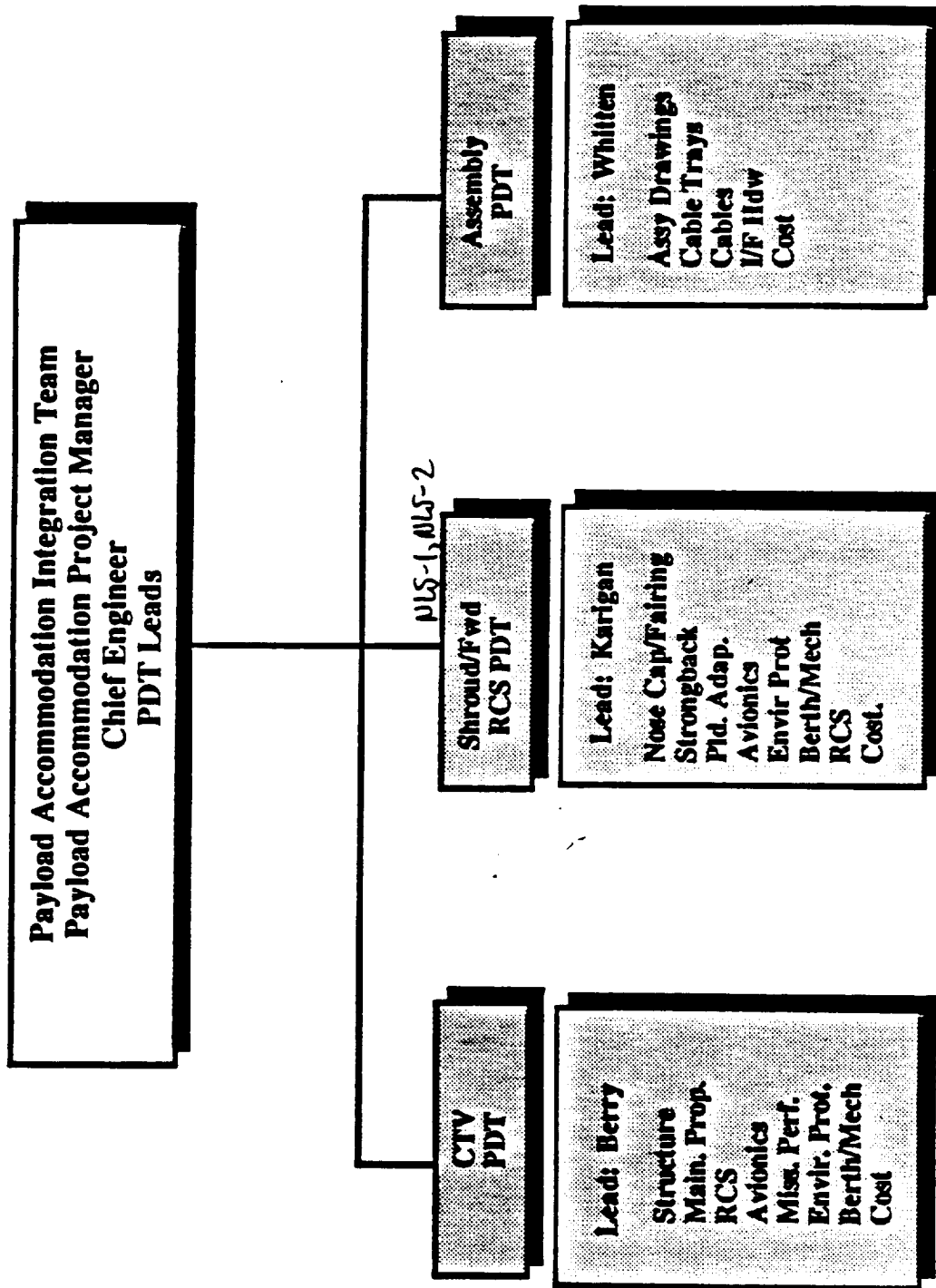


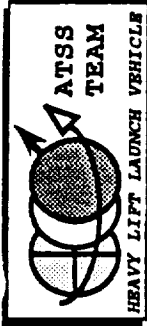
Engine Systems Team



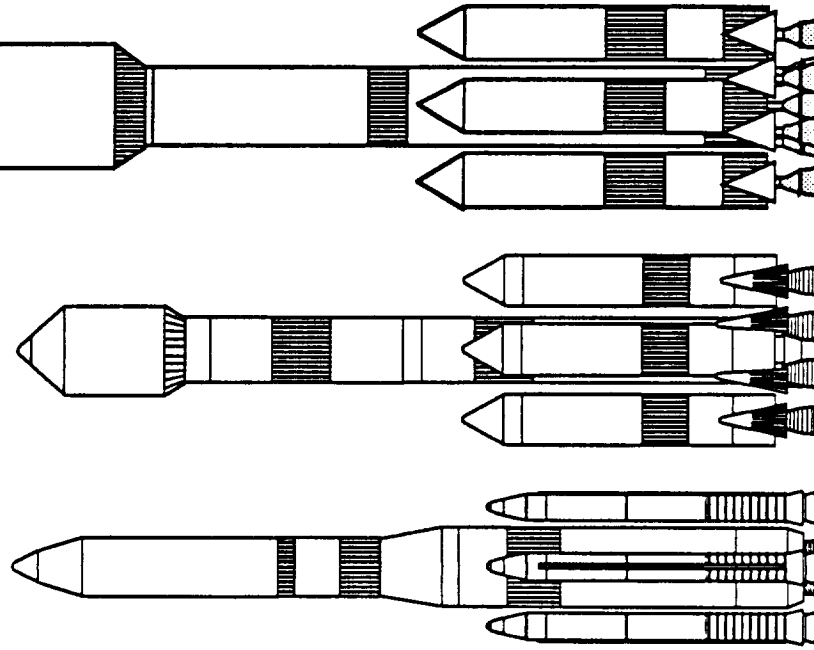


Payload Accommodations Integration Team





TA-2 Concurrent Engineering Meeting #1; 8/5-8/6/92



FLO REQUIREMENTS AFFECTING HLLV DESIGN



J. B. McCurry
205-722-4004



Office of Exploration

First Lunar Outpost Design Activity



- First Lunar Outpost (FLO) is the initial manned component of the Space Exploration Initiative
- FLO design activity is an on-going mission definition and requirements development process that will progress through numerous iterations before final selection of technical approach



First Lunar Outpost Program Strategy

Office of Exploration



- Ensure high mission content
 - Provide lunar science and exploration capabilities exceeding those of Apollo
 - Allow evolution and growth along a number of potential paths as experience and knowledge are gained, and as resources allow
 - 1. *Augustine Report* - "Go as you pay"
 - 2. *Synthesis Group Report* - "Waypoint" Philosophy
- Reduce mission costs
 - Reduce the number of elements and in-space operations required for space transportation
 - Limit number of surface elements required to establish initial lunar capabilities
- Reduce hardware development and deployment times
 - Mission-driven designs; not "technology for technology's sake". New technology only when cost effective.
 - Whenever practical, utilize existing technologies, facilities and infrastructure
 - Reduce number of flights and amount of surface operations required before establishing significant science and exploration capabilities



First Lunar Outpost

Initial Capabilities



Office of Exploration

- Science and exploration strategy parallels program strategy
 - Focused capabilities across many disciplines initially
 - Evolvable to greater emphasis and complexity in one or more areas downstream
- Crew exploration activities include:
 - Investigate the vicinity of the outpost site to a level of detail much greater than that achievable by Apollo.
 - Characterize local resources and demonstrate processes for utilizing them to support lunar operations.
 - Install a suite of observational facilities that surpass existing capabilities in astronomy and space physics.
 - Conduct life science research enabled by unique environment.
- These activities require the following crew support capabilities:
 - Facilities for living, working, and EVA support
 - Local roving (~25 km. from the outpost)
 - Transportation and emplacement of payloads on the lunar surface
- The initially defined mission capability is a crew of four and a lunar surface stay of 45 days (lunar day-night-day).
 - Balance between surface exploration activities and the space transportation and lunar surface infrastructure required.



First Lunar Outpost

Mission Overview



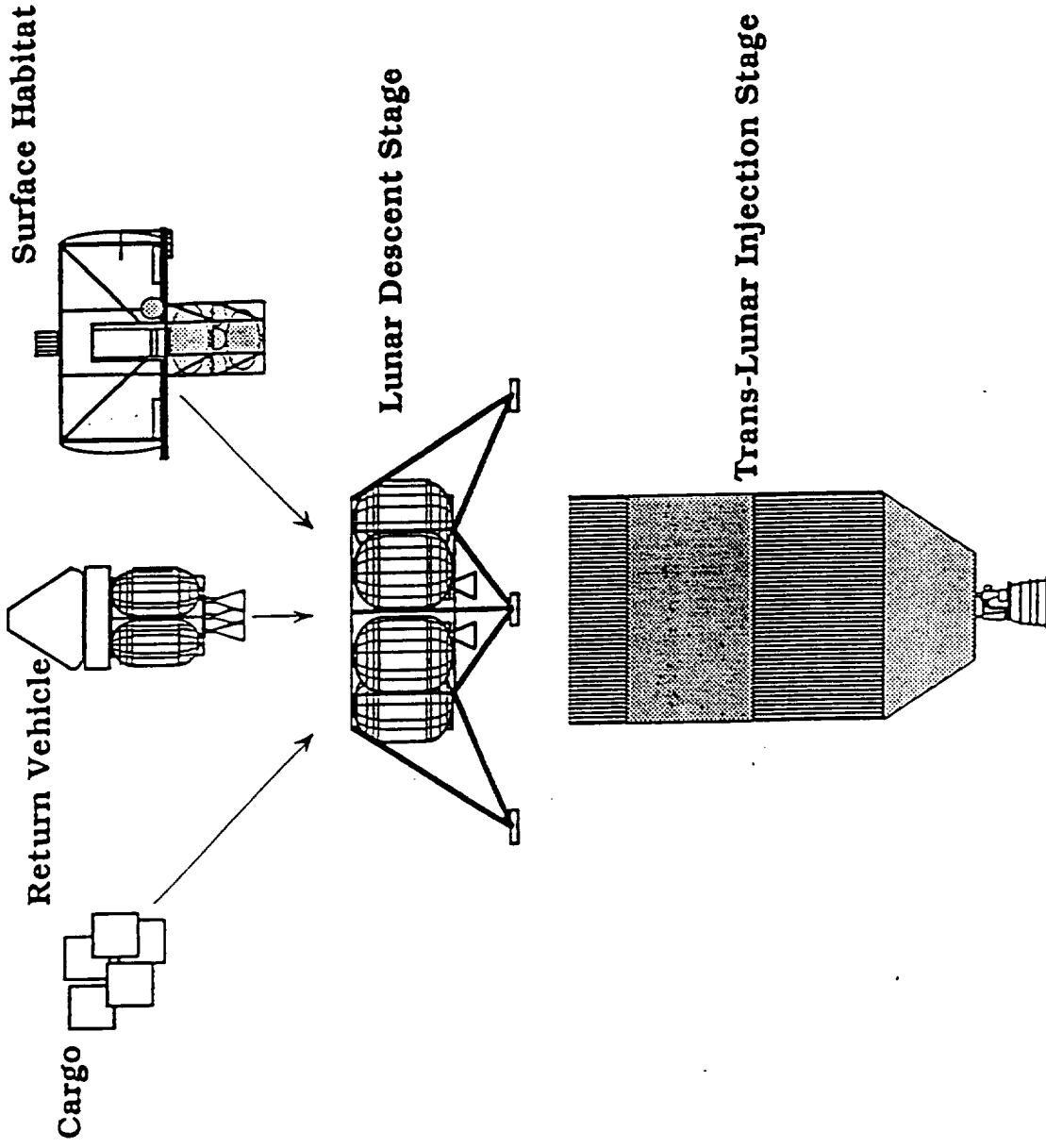
Office of Exploration

- Each flight to the Moon requires a single launch. This greatly simplifies the operations that would be required in a multiple launch scenario.
- No parallel launch vehicle processing is required.
- Shorter on-orbit lifetime for transportation elements is required.
- The need to synchronize multiple launch windows and the trans-lunar injection window is avoided.
- A one-way cargo mission precedes the arrival of the first crew.
- An unmanned lander carrying the lunar habitat and consumables for the first mission lands.
- The habitat is pre-integrated. No construction, emplacement or outfitting is required.
- The habitat "self-deploys" (solar panels, radiators, etc.)
- Confidence of habitability is established prior to the departure of the crew from Earth.
- Piloted flights utilize the "Lunar Direct" mission mode.
- No assets are left in lunar orbit; the entire vehicle descends to the lunar surface.
- This mode allows global lunar access combined with the capability to return to the Earth at any time.
- It allows the development of single crew module only.
- The required surface delivery masses for the piloted vehicle and for the habitat are a good match. This allows commonality between piloted and cargo launch vehicle, trans-lunar injection stage and lunar descent stage.



"Lunar Direct" Transportation System

Office of Exploration





First Lunar Outpost

Mission Overview (cont.)

Office of Exploration



- The piloted flight lands within walking distance of the habitat.
 - The piloted flight also delivers the science payloads and rover
 - The crew unloads the rover and transfers to the outpost. If the rover fails, the crew can walk. No additional deployment or emplacement activities are required immediately
- The initial capability of the habitat is spartan - a "lunar campsite".
 - The habitat does not contain many features which may be desirable for longer duration missions.
 - These capabilities can be added when and if outpost expansion occurs
- After the surface mission is complete, the crew returns to Earth.
 - The crew transitions the habitat to an unmanned mode and returns to the piloted vehicle
 - Lunar liftoff and trans-earth injection are performed by the return stage
 - Crew module performs a direct entry at Earth, similar to Apollo
- Plan to revisit the outpost at ~6 month intervals.
 - Revisit logistics are delivered with the new crew. These are transported to the habitat using the rover



≡ Office of Exploration

HEAVY LIFT LAUNCH VEHICLE REQUIREMENTS



EXPLORATION
PROGRAMS OFFICE

- A SINGLE EARTH LAUNCH VEHICLE SHALL BE UTILIZED FOR EACH FLIGHT TO THE MOON
- FLIGHT ELEMENTS SHALL PROVIDE THE CAPABILITY FOR A FIRST LAUNCH AS EARLY AS 1999
- THE CAPABILITY SHALL BE PROVIDED TO SUPPORT UP TO FOUR HLLV FLIGHTS PER YEAR
- THE FLIGHT SEGMENT (HLLV, TLI STAGE, LANDER) SHALL PROVIDE THE CAPABILITY TO DELIVER 27.5 t FROM THE EARTH'S SURFACE TO THE LUNAR SURFACE.
 - CURRENT ASSESSMENT IS 34-36 t
 - RESULTS IN A 93-96 t REQUIREMENT POST-TLI
 - EQUIVALENT TO 230-250 t TO 100 NMI. CIRCULAR ORBIT



SEI Requirements On The HLLV



Requirements	Trade/Issue	Source	Status
1. The Earth to Moon Transportation System (HLLV; TLI Stage, Lander) Shall Provide The Capability To Emplace 27.5 t (Including 10% Manager's Reserve) On The Lunar Surface In A Single Flight. (Current Assessment Is 34t Of Cargo With Margin Resulting In 23t To TLI)	Red	FLORG # 313	
2. A Single HLLV Shall Be Utilized For Each Flight To The Moon.	Red	FLORG # 246	
3. The HLLV Shall Provide The Capability For Designed Growth To 250t To 220NM.	Red		
4. Flight Elements Shall Provide The Capability To Access Any Lunar Latitude Or Longitude.	Green	FLORG # 149	
5. The HLLV Shall Provide The Capability For Launch As Early As 1999.	Yellow	FLORG # 309	
7. The Capability Shall Be Provided To Support Four (4) Flights Per Year.	Green	FLORG # 307	



Detailed Design Requirements



Requirements	Trade/Issue	Source	Status
Primary			
1. The Usable Shroud Size For Lunar Flights Shall Be $\geq 33 \times 60$ Feet (Goal).	Red	Design G/L	
2. The HLLV Shall Be Designed For No Engine Out On The Core, Boosters Or Upper Stage(s).	Yellow	G/L	
3. The HLLV Shall Be Sized To Provide Launch Capability Any Day During The Lunar Cycle.	Green	G/L	
4. The HLLV Shall Provide The Vehicle Health Monitoring Capability To Provide Notification That An Abort Condition Exists. <u>Launch Escape System (LES) Jettisoned At Shroud Separation (400K Ft).</u>	Green	G/L	
Groundrules/Assumptions			
1. Time Between Cargo And Piloted Flights: 60 Days (Goal).	Green	G/L	
2. <u>No Pad Services Except Fuelling, Checkout And Launch</u>	Yellow	G/L	
3. Maximum Acceleration During Boost Phase: 4g (Goal)	Green	G/L	



Detailed Design Requirements (Concluded)



Requirements	Trade/Issue	Source	Status
4. Capability To Launch From 72° - 108° Azimuth.	Yellow	Design G/L	
5. Maximum Dynamic Pressure During Ascent: 900 PSF	Yellow	G/L	
6. Minimum Lift Off Thrust - To - Weight: 1.2.	Green	G/L	
7. Dry Weight Contingency: 10%	Green	G/L	
8. Ascent Flight Performance Reserve: 1% ΔV	Green	G/L	
9. Jettison Shroud / Nosecap @ 400,000 Ft.	Green	G/L	
10. Lunar Direct Ascent Profile Through 100 NM Earth Orbit (Circular)	Green	G/L	
11. Method Of On Pad Hold Down During Engine Start.	Green	G/L	
12. Primary Avionics Located On TLI Stage.	Green	G/L	



1993

1992

	Jul	Aug	Sep
1			
2			
3			
4			
5			
6			
7			
8			
9			
10			
11			
12			
13			
14			
15			
16			
17			
18			
19			
20			
21			
22			
23			
24			
25			
26			
27			
28			
29			
30			
31			

Feb

Jan

Dec

Nov

Oct

Sep

Aug

Jul

Requirements

- SRR's
- HLLV System Assessment
- HLLV Subelement Derivation

System Concept Definition

- Saturn V Reference
- NLS Reference
- Alternative Trade Studies
- Energia Boosters
- Energia Derived Launch Vehicle
- Clean Shoot Vehicles
- Solids

Subsystems Trades

- Avionics & Power
- Structures
- Thermal
- Propulsion

Systems Analysis

- Layouts
- GN&C
- Flight Mechanics
- Payload Accommodations
- Environmental Impacts
- Design Environments
- Reliability/Risk

Facility Requirements

- Launch Interface
- Manufacturing
- Test

Operations

- Ground Flight**

Programmatics

- **Cost Estimates**
- **Program Schedules**
- **Implementation Plan**
- **Development Approach**

[illegible]



HLLV Technical Interchange Meeting #8

KSC July 15, 1992



<u>Time</u>	<u>Topic</u>	<u>Lead(s)</u>
9:00	KSC Welcome	Blum
9:10	Introduction	Austin
	Potential New Trade Studies	
	Near Term Schedule	
	MSFC Organizational Change For SEI HLLV	
9:30	Status Of Alternate Concepts Definition	
10:00	Trades Studies Status	
	HLLV #1. Launch Vehicle Shroud Size	
	HLLV #2. Alternate Materials On HLLV Concepts	
	HLLV #3. LES Requirements Impacts On HLLV	
	HLLV #4. Feasibility Of Large Launch Vehicle Assy & Opns	
	HLLV #5. Alternate Launch Site Assessment	
	HLLV #6. Evolution To 250t For Mars	
	HLLV #7. Environmental Issues Assessment	
	HLLV #8. Launch Vehicle Drift Assessment	
	HLLV #9. SSME/STME Altitude Start/Restart Assessment Status/Plans	
	HLLV #10. VHM On Vehicles	
	HLLV #11. TVC/Actuator Concept Assessment	
12:00	Lunch	
1:00	Environmental Panel Status/Plans	
1:30	Aerodynamic Panel Status/Plans	
2:00	Acoustics Panel Status/Plans	
2:30	Summary And Wrapup	
	Planning For MSFC/ExPO Coordination Meeting	
	Actions For FLO Engineering Status Review	
	TIM #9	
4:00	Adjourn	



Trade Studies

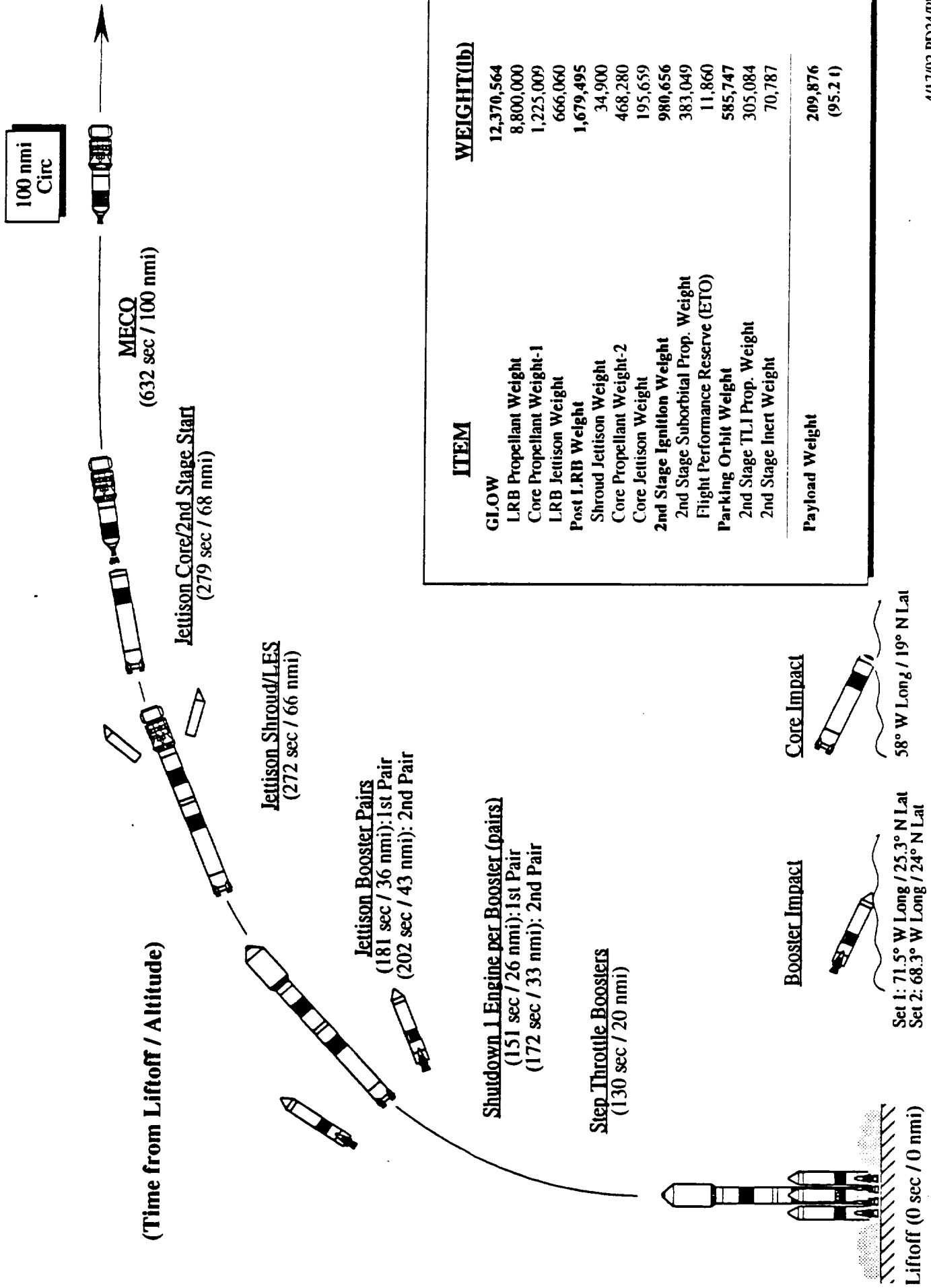


1. Launch Vehicle Shroud Size
 - Cylindrical Diameter And Length Requirements With Space Transit Subbeam
 - Nosecap Bluntness Requirements With Space Transit Subbeam
2. Alternate Materials On HLLV Concepts
 - Al-Li
 - Composites
3. LES Requirements Impacts On HLLV
 -
 -
4. Feasibility Of Large Launch Vehicle Assy & Opns
 - Manufacturing
 - Assembly
 - Launch Operations
5. Alternate Launch Site Assessment (Support The Operations Subteam)
 - Baikonr (CIS)
 - Cape York (Australia)
 - Others (Including Off Shore)
6. Evolution To 250t For Mars
7. Environmental Issues Assessment
 - Manufacture
 - Test
 - Launch
8. Launch Vehicle Drift Assessment
9. SSME/STME Altitude Start/Restart Assessment
10. VHM On Vehicles
11. TVC/Actuator Concept Assessment
 - EMA's
 - EHA's
 - Hydraulics
12. Minimization Of New Infrastructure (With The Operations Subteam)
 - Resize Elements To Stay Within Current Facility Constraints
 - Multiple Launch Lunar Mission
 - Reduce Mars Evolution Requirements

Internal Trades

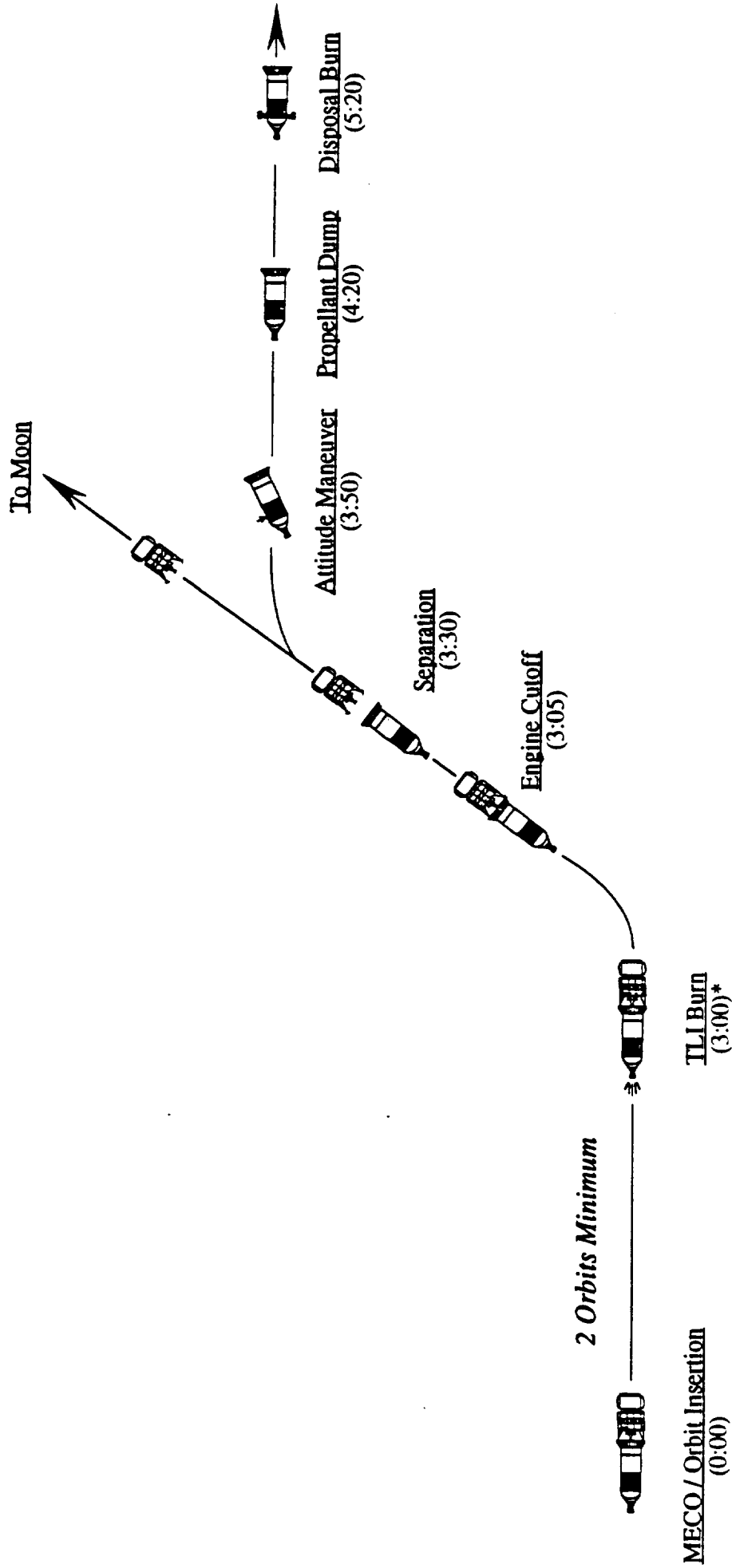
Lunar Mission Profile

Ascent Phase



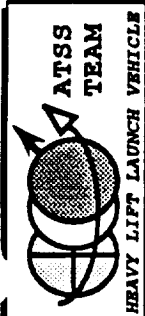
Lunar Mission Profile

Orbital Phase

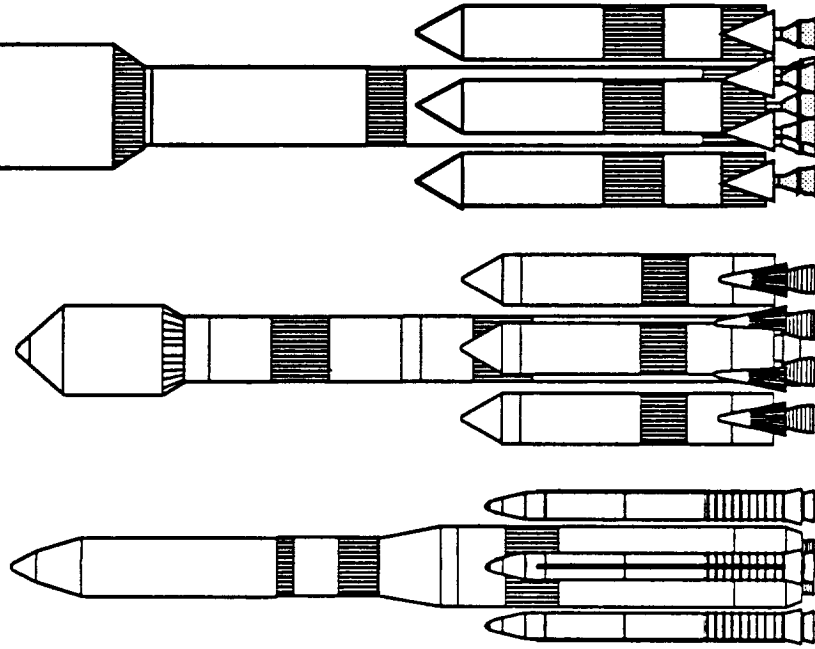


(Hr:Min from Orbit Insertion)
Nominal

* Determined by Mission Window (Max = 6:45)



TA-2 Concurrent Engineering Meeting #1; 8/5-8/6/92



FIRST LUNAR OUTPOST REFERENCE VEHICLE CONFIGURATIONS

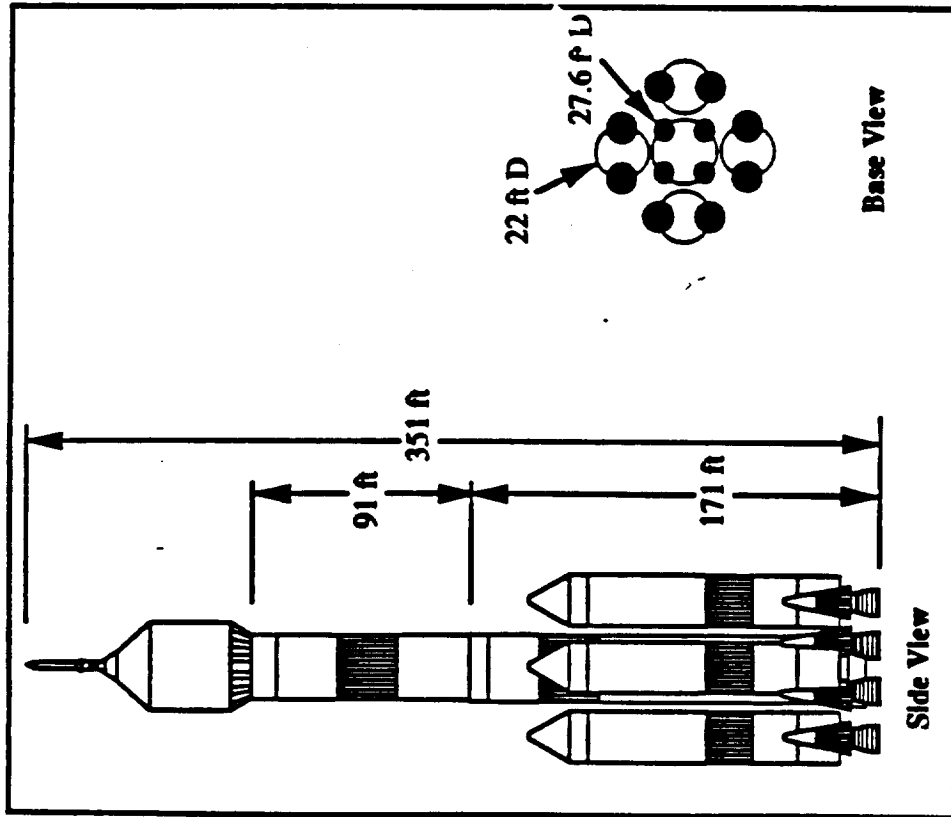


J. B. McCurry
713-333-6579



NLS Derived HLLV w/ 4 LOX/RP Boosters

Single Launch - Piloted



Shroud - Size: 38 x 47 ft
Mass: 23,860 lb

- Notes:**
- F-1A & STME are 75% Step Throttleable
 - Use Throttle/Booster Engine Shutdown for Loads
 - Max G = 4.0 / Max q = 900 psf
 - 108° Launch Azimuth

Payload: 210 klb (95 t) / 586 klb (267 t)
Final Position: TL/LEO Cutoff
GLOW: 12.4 Mib
Engine Out: None

BOOSTER:

Number/Type: 4/New
Inert Mass: 166.5 klb
Propellant Mass: 2.2 Mib
Propellant Type: LOX/RP
Engine Type/#: F-1A/2
Vac /SL Thrust (Ea): 2.02/1.8 Mib
Vac ISP: 303.1 s

CORE:

Inert Mass: 195.7 klb
Propellant Mass: 1.69 Mib
Propellant Type: LOX/LH2
Engine Type/#: STME/4
Vac /SL Thrust (Ea): 650/551 klb
Vac /SL ISP: 428.5/365 s

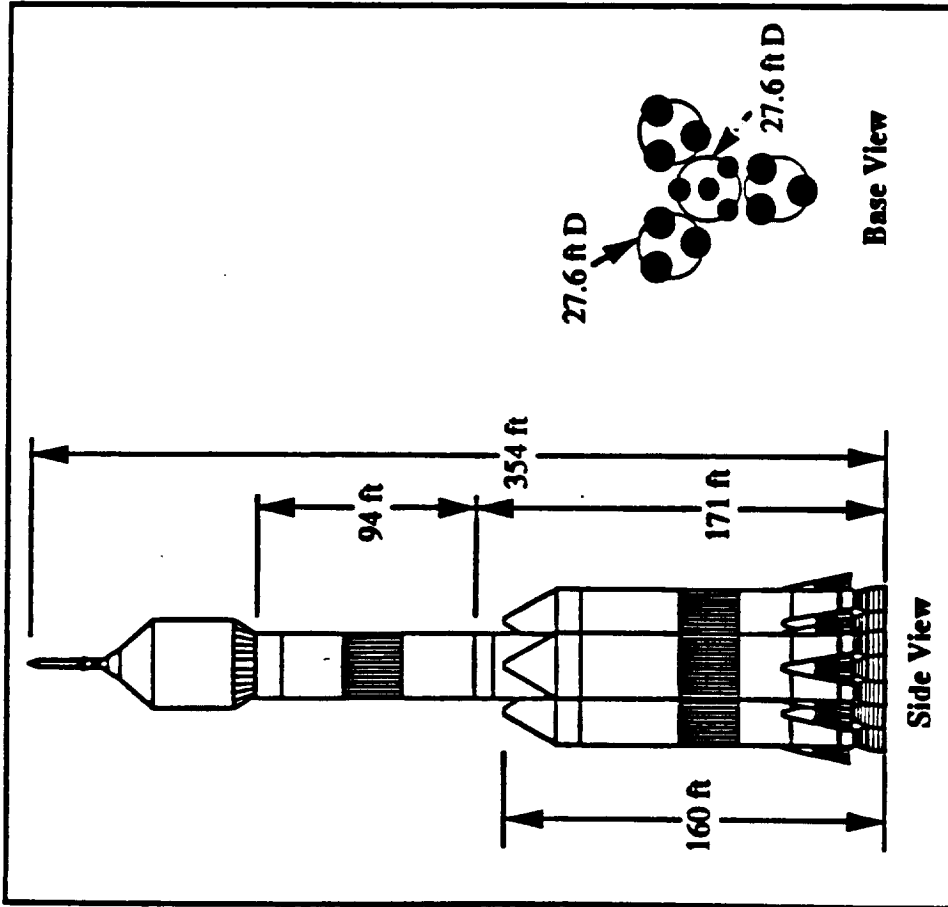
TLL Stage:

Inert Mass: 70.8 klb
Propellant Mass: 700 klb
Propellant Type: LOX/LH2
Engine Type/#: SSME/1
Vac Thrust (Ea): 489.9 klb
Vac ISP: 452.4 s



NLS Derived HLLV w/ 3 LOX/RP Boosters

Single Launch - Piloted



Shroud - Size:
Mass:

- Notes:
- F-1A & STME are 75% Step Throttleable
 - Use Throttle/Booster Engine Shutdown for Loads
 - Max G = 4.0 / Max q = 900 psf
 - 108° Launch Azimuth

Payload: 208 klb (94t) / 615 klb (280 t)
Final Position: TLM/LEO Cutoff
GLOW: 13.6 Mib
Engine Out: None

BOOSTER:

Number/Type: 3/New
Inert Mass: 242.3 klb
Propellant Mass: 3.3 Mib
Propellant Type: LOX/RP
Engine Type/#: F-1A/3
Vac /SL Thrust (Ea): 2.02/1.8 Mib
Vac ISP: 303.1 s

CORE:

Inert Mass: 196.5 klb
Propellant Mass: 1.69 Mib
Propellant Type: LOX/LH2
Engine Type/#: STME/4
Vac /SL Thrust (Ea): 650/551 klb
Vac /SL ISP: 428.5/365 s

TLL Stage:

Inert Mass: 70.8 klb
Propellant Mass: 700 klb
Propellant Type: LOX/LH2
Engine Type/#: STME(65:1)/1
Vac Thrust (Ea): 659 klb
Vac ISP: 434.6 s

Booster 1 Concept

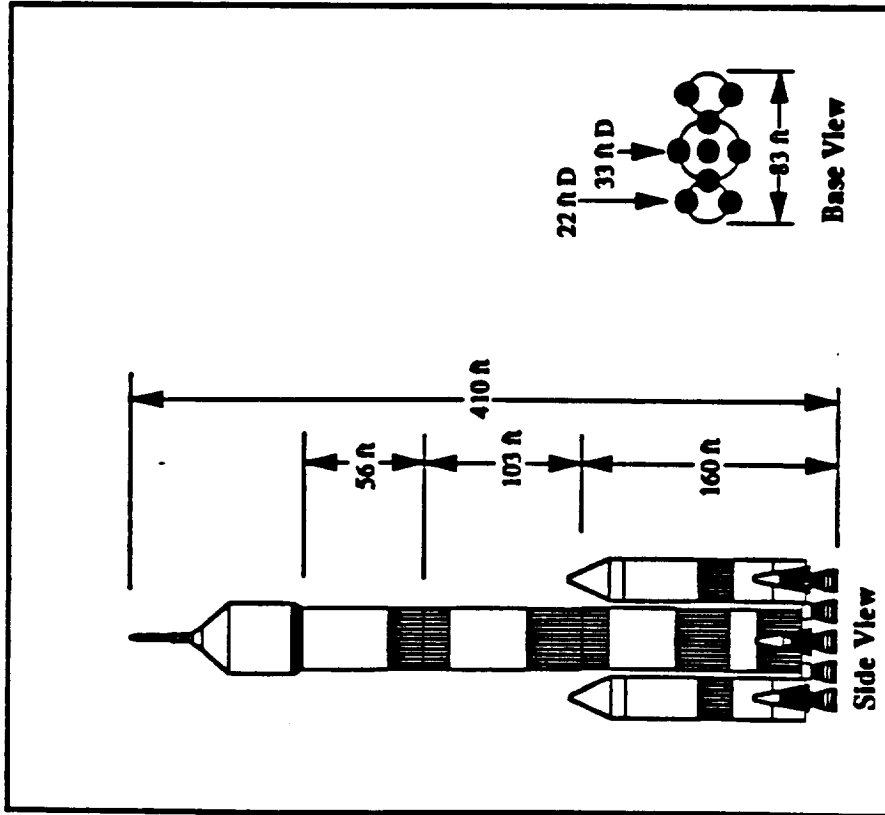


Saturn V Derived HLLV w/ 2 LOX/RP Boosters

Single Launch - Piloted



EXPLORATION
PROGRAMS OFFICE



Payload: 215 klb (98 t) / 561 klb (254 t)
Final Position: TL/LEO Cutoff
GLOW: 13.3 Mlb
Engine Out: None

BOOSTER:

Number/Type: 2/New
Inert Mass: 167 klb
Propellant Mass: 2.2 Mlb
Propellant Type: LOX/RP
Engine Type/#: F-1A/2
Vac /SL Thrust (Ea): 2.02/1.8 Mlb
Vac ISP: 303.1 s

CORE S-IC:

Inert Mass: 461 klb
Propellant Mass: 6.0 Mlb
Propellant Type: LOX/RP
Engine Type/#: F-1A/5
Vac /SL Thrust (Ea): 2.02/1.8 Mlb
Vac ISP: 303.1 s

CORE S-II:

Inert Mass: 134 klb
Propellant Mass: 1.4 Mlb
Propellant Type: LOX/LH₂
Engine Type/#: J-2S/6
Vac /SL Thrust (Ea): 265 klb
Vac ISP: 436 s

TLL Stage:

Inert Mass: 47 klb
Propellant Mass: 296 klb
Propellant Type: LOX/LH₂
Engine Type/#: J-2S/1
Vac Thrust (Ea): 265 klb
Vac ISP: 436 s

33 x 47 ft
23,860 lb

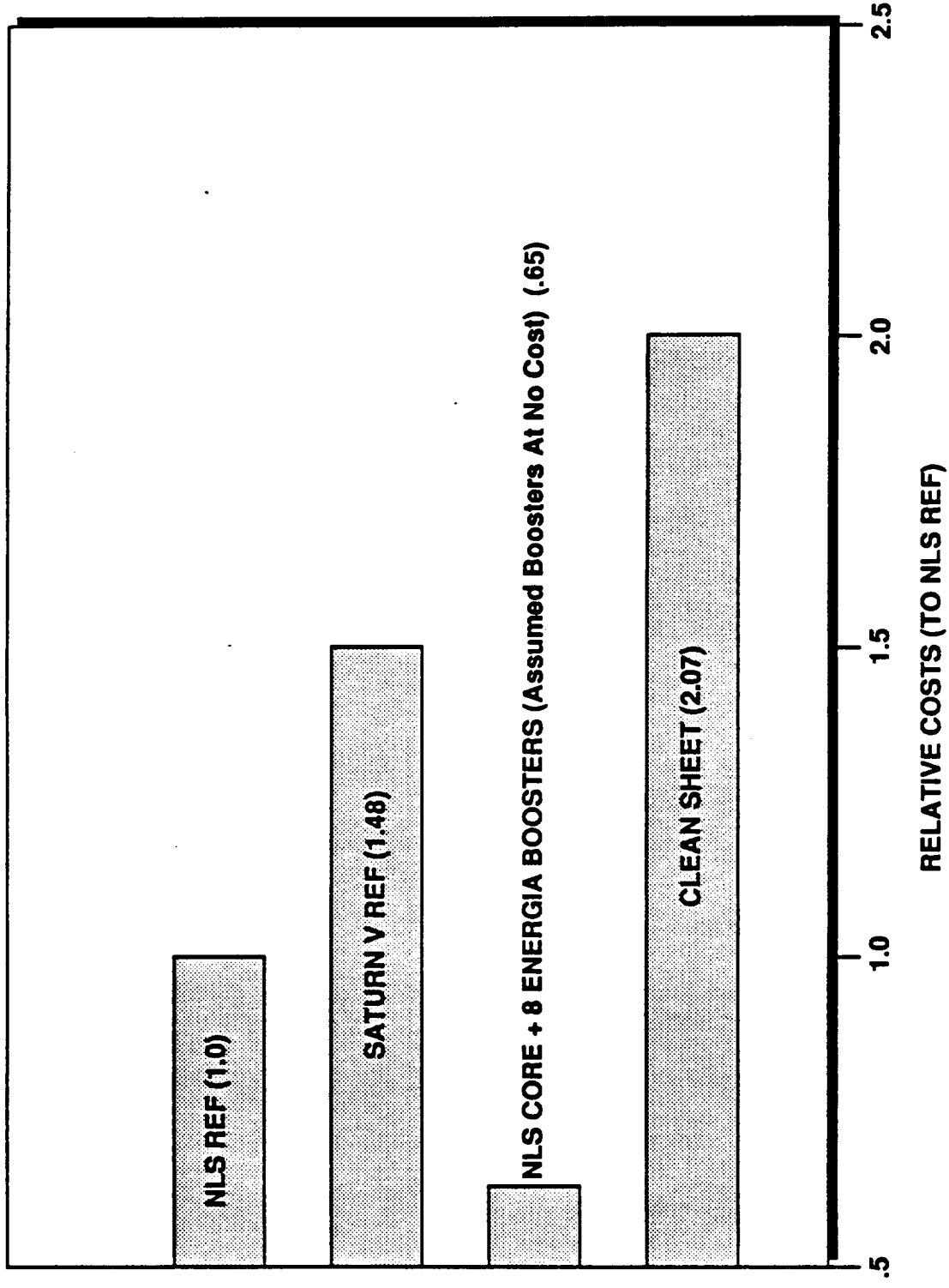
Shroud - Size:

Mass:

- Notes:
- F-1A's are 75% Step Throttleable
 - Use Throttle For Loads
 - Max G = 4.0 / Max q = 900 psf
 - 72° Launch Azimuth

COST COMPARISON - SEI VEHICLES

DDT&E COSTS



* NOTE: Costs do not include facilities

SSME Upper Stage Use

Potential Issues

- **Current SSME Start**
 - **Tank-Head Start**
 - Liquid at Pump Inlets
 - High Positive Pressure at Pump Inlets
 - **Known Turbine Conditions (Moisture/Ice/Turbine Physical Environment)**
 - Purged
 - **Turbine Start with Hardware at Conditions Around Ambient Sea Level**
 - Sufficient Hardware Enthalpy for Bootstrap Start
- **Altitude and Orbital Starts and Restarts**
 - **Different Pump Inlet Conditions**
 - Second Stage/TLI Stage Pressures
 - Staging
 - Zero g
 - **Turbine Environment**
 - **Changed Conditions for Restart**
 - **Turbine Start**
 - **Hardware Enthalpy May be Very Low**

SSME Upper Stage Use Altitude Start Conclusions

41

- **Preburner Valves Need to be Sequenced to Higher Positions and Modified Timings to Accommodate Lower Inlet Pressure**
- **Initial Bootstrap Rate Reduced From Current Start**
- **Time to Reach Mainstage Not Affected**

ALTERNATE CONFIGURATION OPTIONS

SEI LAUNCH VEHICLES

- **ENERGIA HARDWARE / FACILITIES**
- **ASRM OPTIONS**
- **RSRM OPTIONS**

HISTORY OF ZENIT / ENERGIA

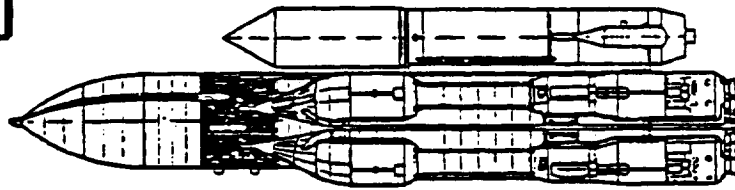
Zenit



LAUNCH RECORD

4-13-85 Suborbital Test Fit
 6-21-85 Suborbital Test Fit
 10-22-85 Kosmos 1967
 12-28-85 Kosmos 1714
 7-30-86 Kosmos 1767
 10-22-86 Kosmos 1786
 2-14-87 Kosmos 1820
 3-18-87 Kosmos 1833
 5-13-87 Kosmos 1844
 8-1-87 Kosmos 1871
 8-28-87 Kosmos 1873
 5-15-88 Kosmos 1943
 11-23-88 Kosmos 1980
 5-23-90 Kosmos 2082
 10-4-90 ** 1st Stg Failure **
 Aug 91 ** 2nd Stg Failure **
 2-5-92 ** 2nd Stg Failure **

Energia



LAUNCH RECORD

5-15-87 Energia 1st Fit
 ** Mockup Payload Failed To
 Insert Itself Into Orbit **
 11-15-88 Buran 1st Fit

TOTAL FLIGHTS:

2 Test, 12 Successes, 3 Failures

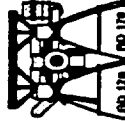
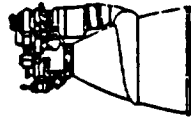
2 Successes

- First Totally New Soviet Launch Vehicle In 20 Years
- LOX / Kerosene 1st & 2nd Stage
- RD-170 1st Stage Engine; Largest Existing Engine And Qualified For 10 Reuses
- First Stage Nearly Identical To Energia Booster (Synergism)
- 30 kibs Payload To LEO (51.6° Inclination)
- Three Stage Version IOC 1993
- Two Launch Pads - One Currently Out Of Service After 1990 1st Stage Failure

- Energia Program Initiated In Early 1970's After Failure Of N-1 Moon Rocket Program
- Uses Zenit First Stage As Booster
- LOX / LH2 Monolithic Core (w/ Engines)
- ~200 kibs Payload To LEO (51.6° Inclination)
- Three Launch Pads - Two Capable Of Launching Buran

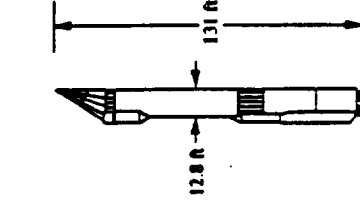
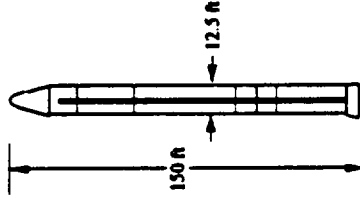
Comparison of U.S. / CIS Hardware

Liquid Engine Comparison



	P-1A	RD-170
Thrust (lb)		
VAC	2,020,500	1,776,892
SL	1,800,000	1,631,316
ISP (s)		
VAC	304.2	336.0
SL	271.0	308.5
Mixture Ratio	2.27:1	2.63:1
Chamber Pressure (psia)	1,161	3,556
Expansion Ratio	16:1	36.8:1
Length (ft)	18.4	13.2
Nozzle Exit Dia (ft)	12	4.7 (each nozzle)
Weight (lb), inc TVC	20,000 (est)	23,507 (est)
Throttle	Option: Step to 75%	50% 100% 47%

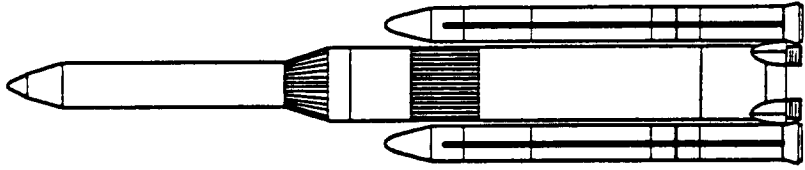
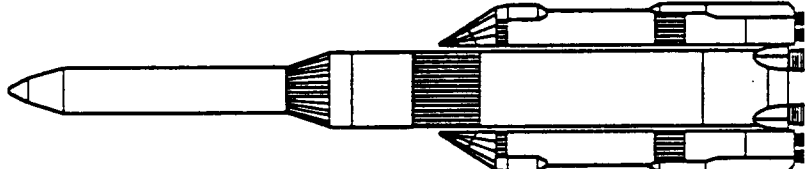
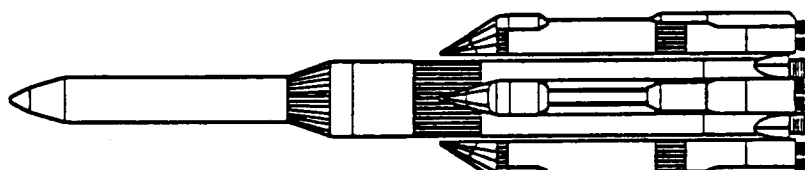
Booster Comparison



Notes:
Derived from Zenit
Launch Vehicle

	ASRM	Energia Booster
Total Thrust - SL	3.2 Milb	1.63 Milb
Inert Weight	176 klb	78 klb
Propellant Type	HTPB	LOX/RP
Propellant Mass (usable)	1.21 Milb	705 klb
# Engines / Type	1 / Solid	1 / RD-170 (4 Nozzles)
Reusable	Yes	Possible (10 flights)

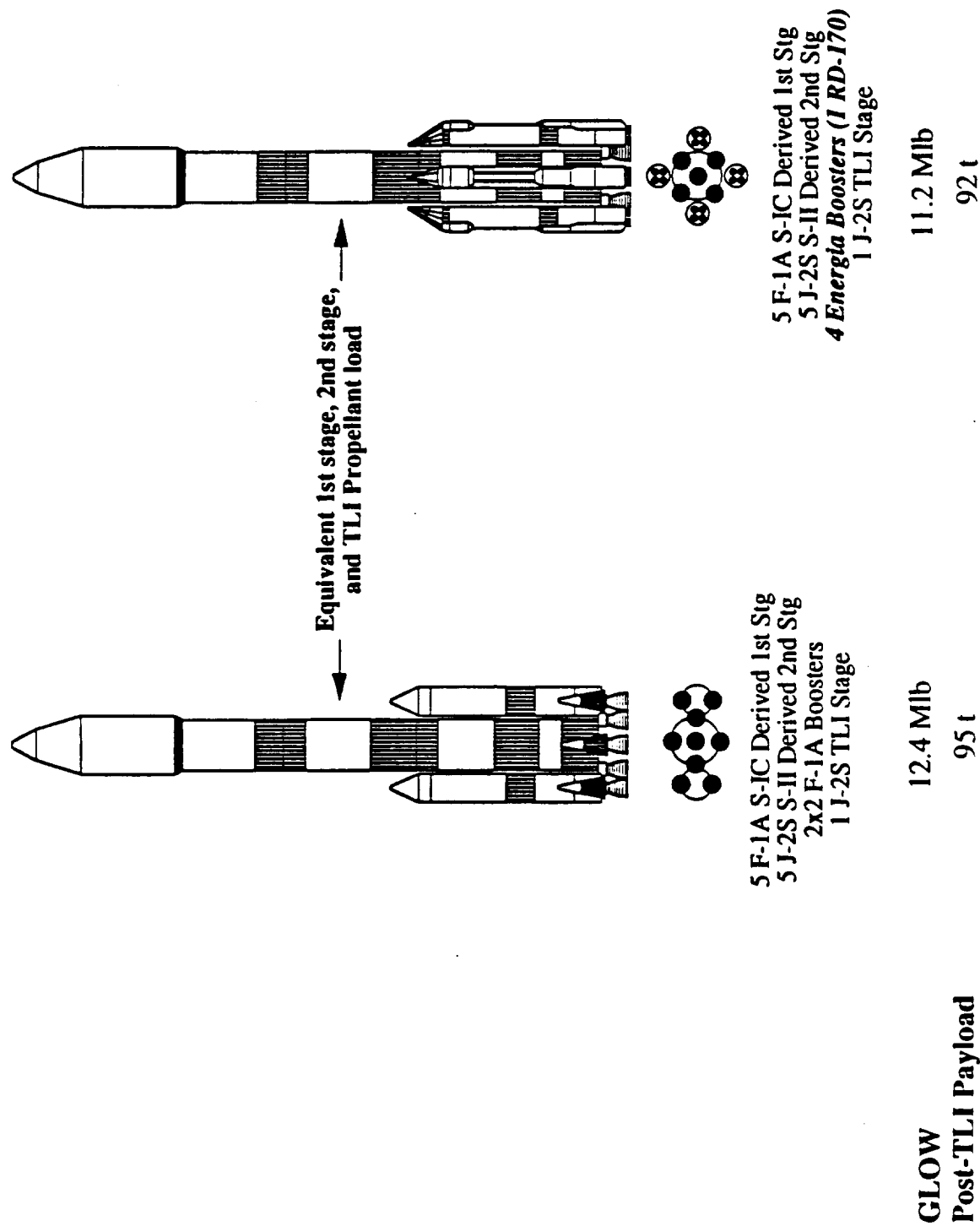
NLS w/ Energia Strap-On Boosters

	NLS (Reference)	2 Booster	4 Booster
			
Payload to LEO	146 klb	133 klb ($\Delta = -8.9\%$)	240 klb ($\Delta = -64\%$)
GLOW	4.8 Milb	3.6 Milb	5.25 Milb

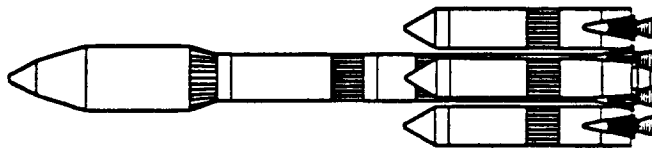
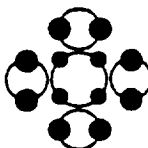
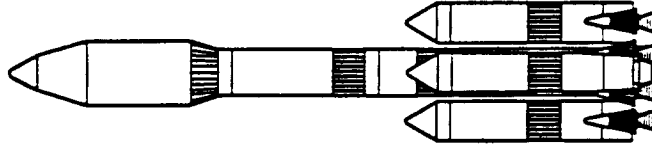
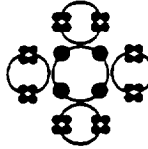
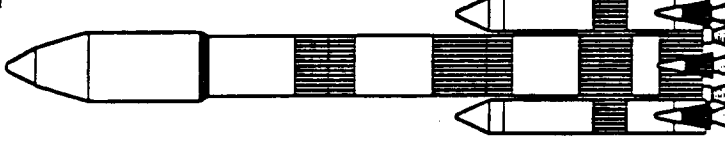
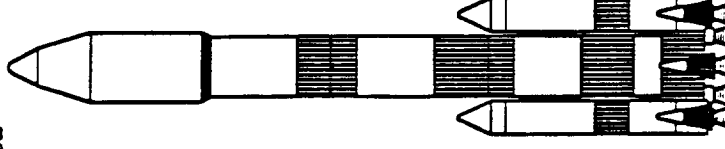
with 250 E/O

- Notes:
- 80 x 150 nmi @ 28.5°
 - One Core Engine Out
 - Max g = 4
 - Max q ≤ 900 psf
 - Wp Core = 1.69 Milb

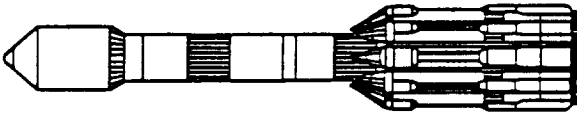
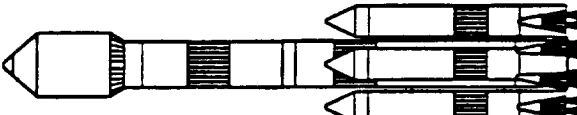
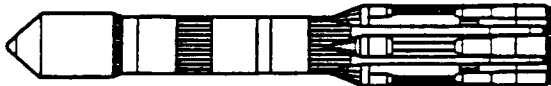
Saturn V w/ Energia Strap-On Boosters



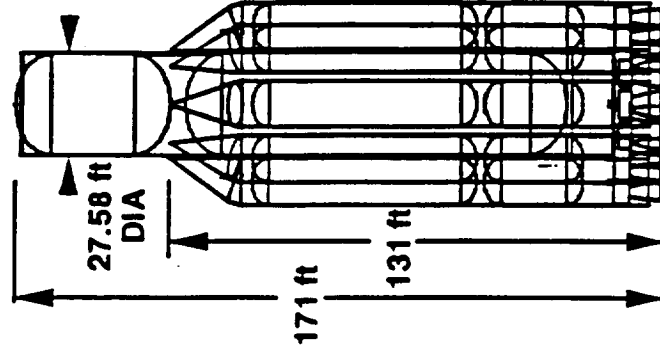
Alternate Engine Configurations

<i>NLS Derived</i>		<i>Saturn V Derived</i>	
			
NLS Core (4 STME) 4x2 F-1A Boost 1 SSME TLI Stage	NLS Core 4x2 RD-170 Boost 1 SSME TLI Stage		
			
			5 F-1A (Str. S-IC): 1st Stg 5 RD-170 S-IC Derived 5 J-2S (Str. S-II): 2nd Stg 5 J-2S S-II Derived 2x2 F-1A Boost 2x2 RD-170 Boost 1 J-2S TLI Stage 1 J-2S TLI Stage
GLOW 12.4 Milb	12.4 Milb	12.4 Milb	12.2 Milb
Post-TLI P/L 95 t	95 t	95 t	109 t

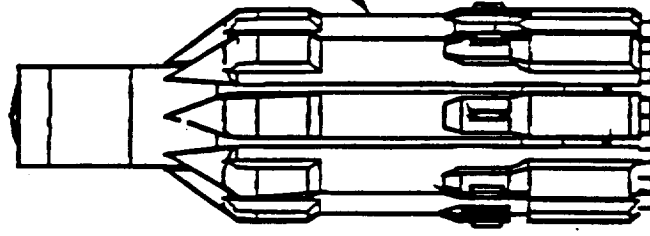
SEI Launch Vehicle Specifications

			
Payload (Post TLI)	<i>(100% CASE)</i> A2.2.105kg ~ KSC: 93 t Baikonur: 85 t	KSC: 103 t	KSC: ?? t Baikonur: ?? t
GLOW	9.3 Mlb	10.9 Mlb	?? Mlb
Core Description	1.69 Mlb NLS Derived Stage 4 STME	1.69 Mlb NLS Derived Stage 4 STME	2.4 Mlb / 33 ft Dia. Stage 4 STME
Booster Description	8 Energia Boosters 1 RD-170 Engine Each	4 x 1.8 Mlb LOX/RP Stage 2 RD-170 Engines Each	6 Energia Boosters 1 RD-170 Engine Each
TLI Stage Description	850 klb LOX/LH2 Stage 1 SSME	700 klb LOX/LH2 Stage 1 SSME	?? klb LOX/LH2 Stage 1 SSME

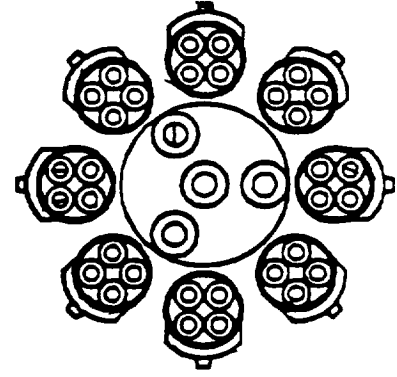
NLS Core With Eight Energia Boosters



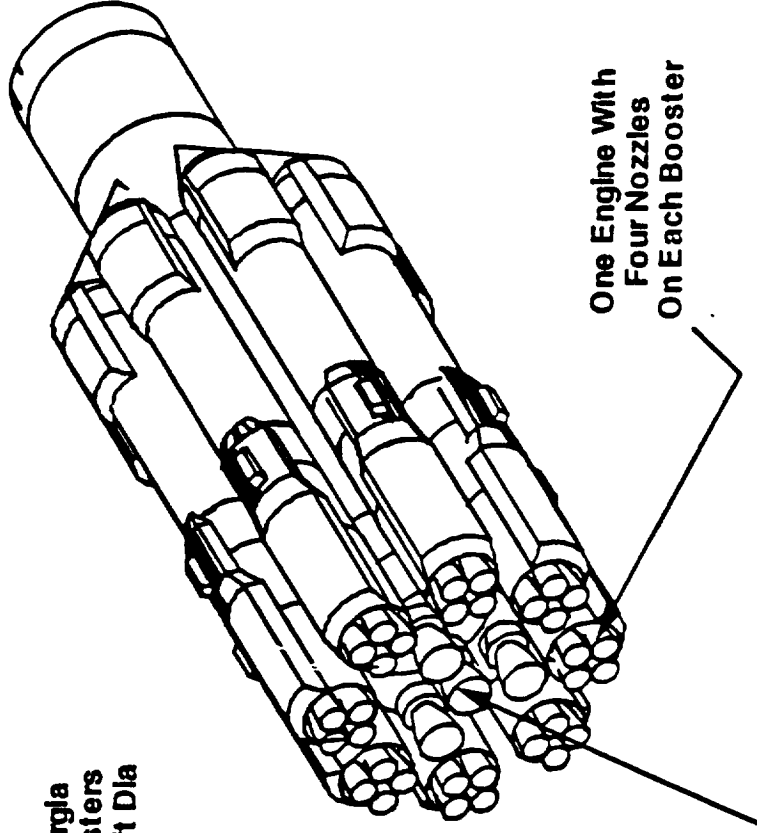
Energia
Boosters
12.8 ft Dia



Side Views



End View
(enlarged)



One Engine With
Four Nozzles
On Each Booster

4 STME Engines
On Core Stage

Issues / Concerns with Using CIS Hardware / Facilities

General

- Reliance on a Politically Unstable Foreign Source for Flight Critical Hardware / Facilities
- Limits of CIS Liability
- Language Difference (Technical Translation)
- Technology Transfer
- Establishing a Cost for Hardware / Services

Hardware

- Economic Impacts on U.S. Industry
- Testing and Qualification of Hardware
- Reliance on a Two Nation Consortium for Hardware (Boosters - Ukraine, Engines - Russia)
- Reliability
- Handling, Checkout, and Maintenance Requirements
- RD-170 is Complex Engine
 - Stage Combustion Cycle
 - High Chamber Pressure (3500 psi)
 - Uses Hypergolic Charge for Ignition (Safing Concern)
- Verification of Vendor Quality
- Interface with U.S. Stages and Payloads (Structural, Thermal, Propulsion, and Avionics/Power)

Facilities (Balkonur Cosmodrome)

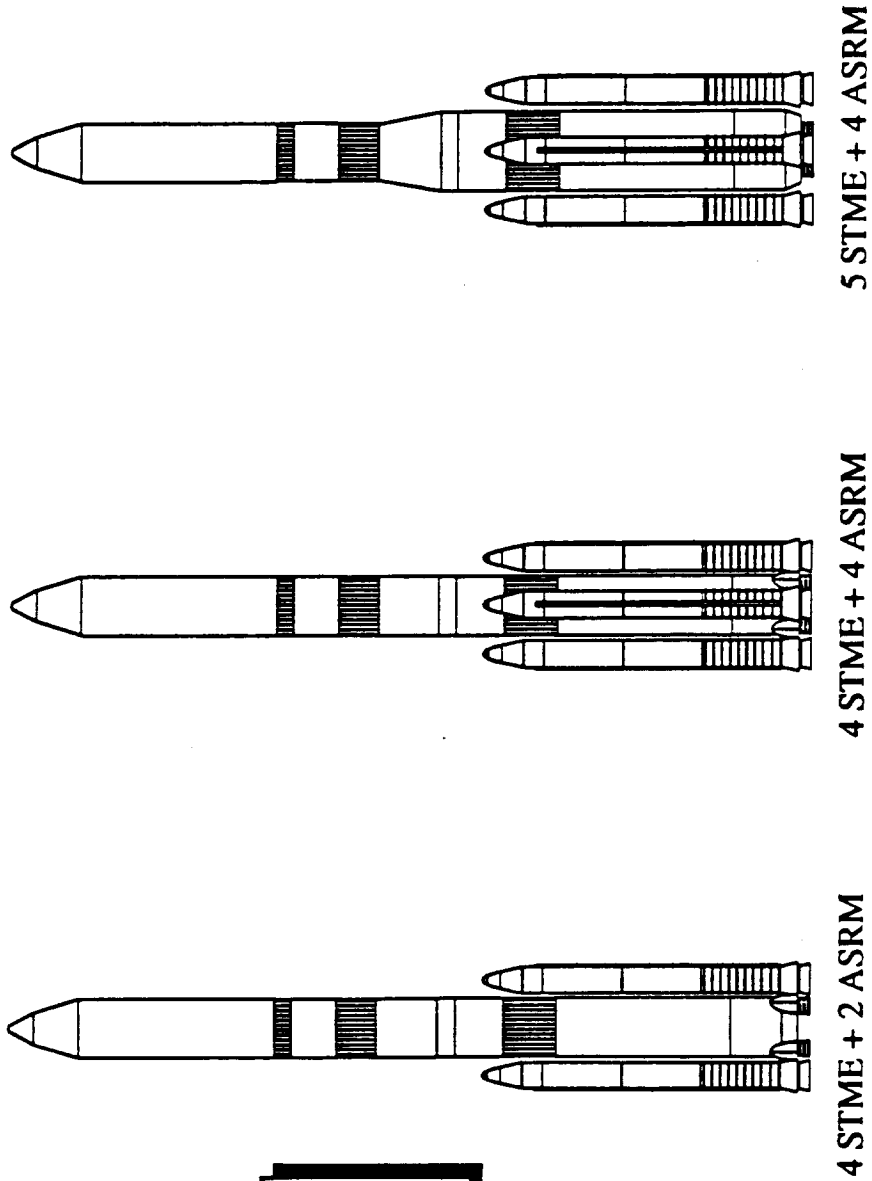
- Stage and Payload Processing Philosophy Differences (e.g., Horizontal vs. Vertical Processing)
- Logistics Associated with Shipping U.S. Stages/Payloads to CIS
- Ground Support Equipment Interfaces
- Launch Site Availability (Russia Does Not Control Balkonur)
- Mission Constraints due to Launch Site Location and Overflight Constraints
 - Southerly Launch Azimuth Constraint = 60° (51.6° Inc Orbit) - China Overflight
 - Northerly Launch Azimuth Determines Launch Window
 - 37° Azimuth = 2 Hour Window (Design Condition)
 - 5° Azimuth (≈ Polar) = 4 Hour Window (= KSC Window)
- Possible Arctic Abort Scenarios
- Possible Stage Disposal Constraints

SEI Launch Vehicles

Possible ASRM Options

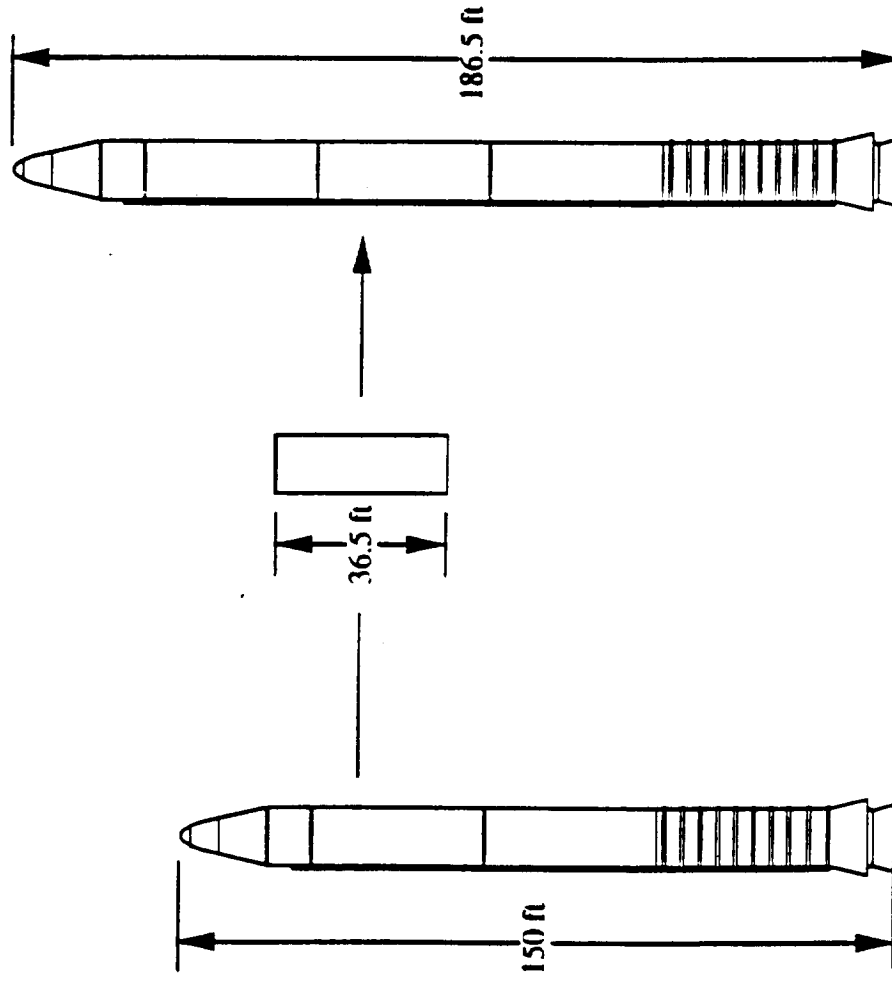
Note:

- Single Launch
- 4 G / q=900 psf constraint
- STME 75% Step Throttle
- 25 x 90 ft Payload Envelope
- 300-500 klb Class LOX/LH2 TLI Stage (Sub-Orbital Burn)



Core Diameter	27.6 ft	27.6 ft	37 ft
Core Prop. Load	1.69 Mlb	1.69 Mlb	3.0 Mlb
Post TLI Payload	35 t	45 t	≈ 65 t

ASRM w/ Extra Segment



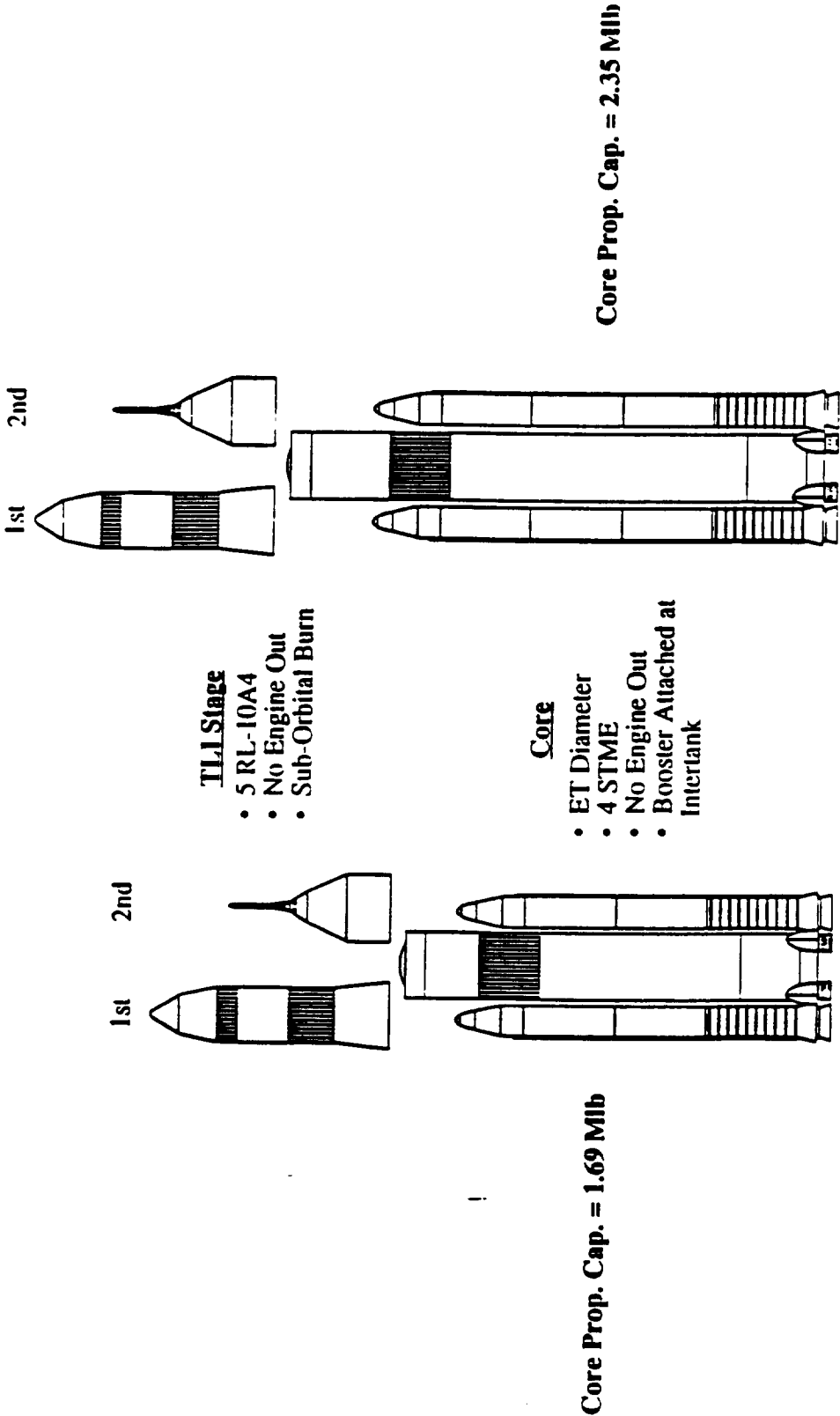
Two Options: Enlarged Throat Area
Nominal Throat Area

Total Weight *	1.3 Milb	1.8 Milb
----------------	----------	----------

* Does not include nosecone, chutes, sep motors, TVC system, etc. Also does not account for any redesign changes that may be required if an extra segment is added

NLS Derived Lunar Vehicles

2 Launch Rendezvous and Dock



NLS Core w/2 ASRM

Stretched NLS Core w/2 Extra Segment ASRM

GLOW	5.0 Milb	6.8 Milb
Post-TL1 Payload	70 mT	102 mT
	Enlarged Throat	Nominal Throat

Four-Segment ASRM

Four-segment ASRM with redesigned nozzle (Case I)

Characteristics

- Standard ASRM plus additional center segment
- Redesigned nozzle with enlarged throat to maintain standard ASRM operating pressure
- Burn time 145 sec

Observations

- Erosive burning likely
- Possible combustion instability problems due to increased motor L/D
- Redesigned nozzle throat area larger than grain port area

- GRAIN REDESIGN REQUIRED

Four-segment ASRM with increased operating pressure (Case II)

Characteristics

- Standard ASRM plus additional center segment
- Burn time 115 sec

Observations

- Erosive burning likely
- Possible combustion instability problems due to increased motor L/D
- Nozzle erosion problems
- Modification of propellant chemistry maybe required to reduce operating pressure and increase burn time
- Operating pressure may exceed motor case design limits

- CASE REDESIGN REQUIRED

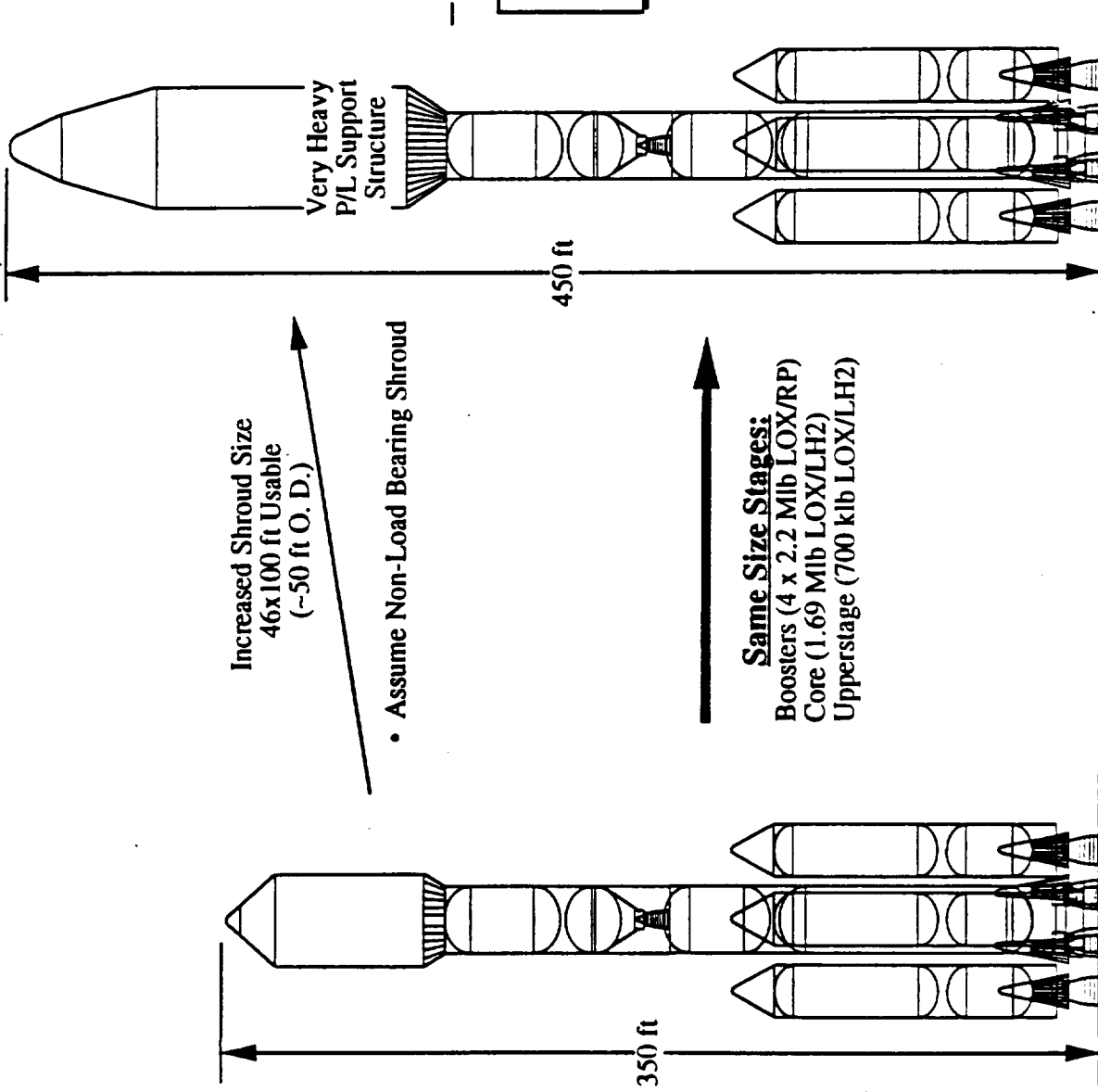
* Total Impulse: Four-Segment ASRM = 4.38×10^8 lbf-sec; Standard ASRM = 3.24×10^8 lbf-sec

Issues With 4 + Solid Booster Configurations

- New Launch Pad Required (Exhaust Trench)
- New Crawler Required
- New Mobile Launch Platform Required
- Crawlerway Modifications Possible
- VAB MLP Mount Capability is Questionable (Sunk Into Foundation)
- Stacking Concerns
 - "Tilt - In" Of Multiple Boosters
 - Different than LRB's Since Are Hazardous and Weigh 1.25 Milb +
- Grain Redesign May Be Required to Reduce Dynamic Pressure
- Local Acidification Due to Al in Propellant

NLS Derived SEI Vehicles

Mars Payload Accomodations



Lunar Vehicle

Mars Vehicle

Loads

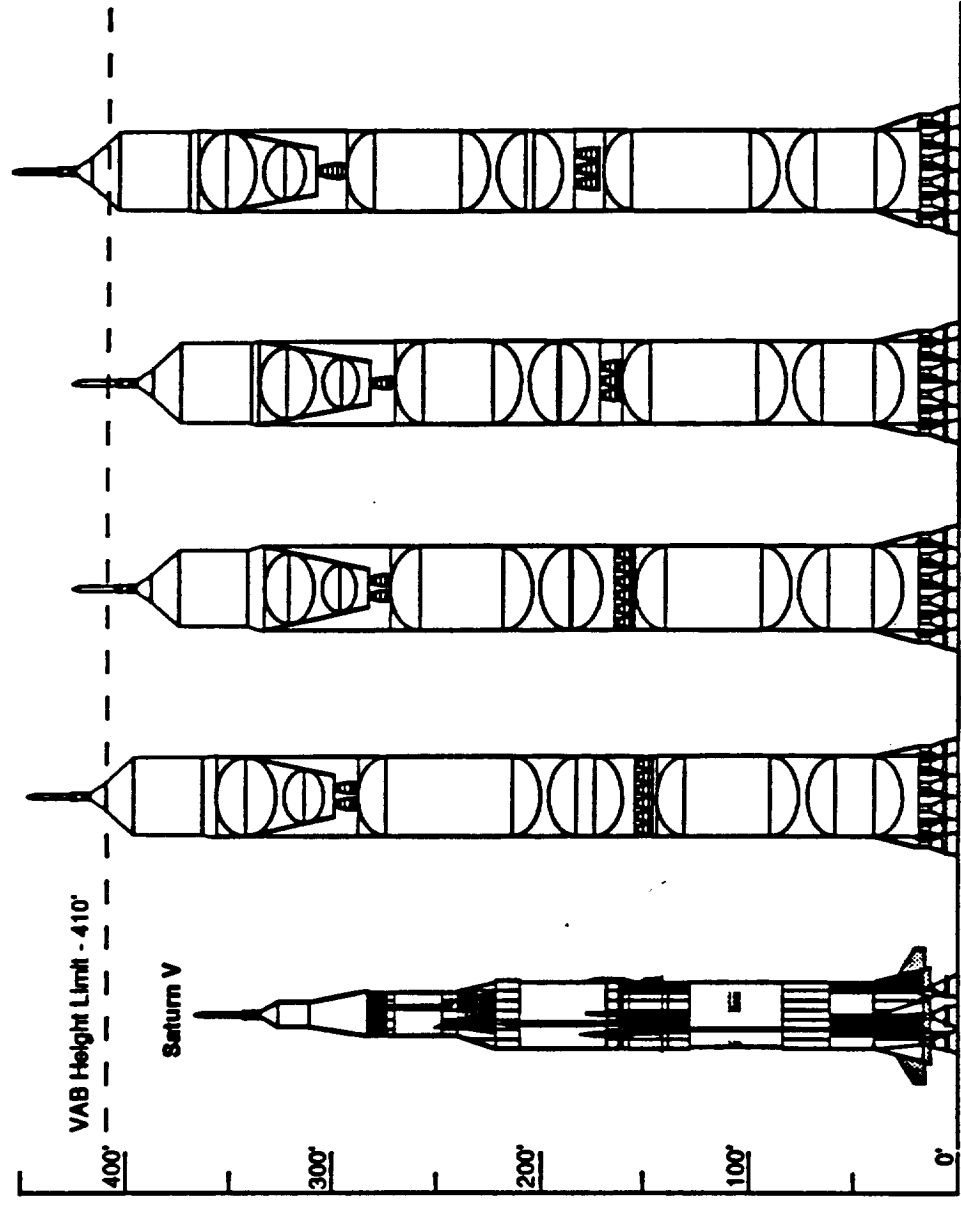
- Larger Shroud Planform Area
 - Larger Bending Moment
- Cp Moves Forward
 - Larger Aerodynamic Instability
 - Larger Bending Moment Arm
- Larger Shroud+P/L Mass
 - Increased Axial Force
- Large Neck Down
 - Increased Bending Moment

Structures

- Ring Frame/Stringer Construction
- Tank Skins Sized Primarily by Pressure and Buckling
- Stringer Geometry Sized by Buckling
- Upperstage tanks will have to be strengthened
- Core LOX tank will have to be strengthened
- Intertank/stages will have to be strengthened (larger T panels and rings)
- Core LH2 Tank will have to be strengthened
- Boosters remain largely unaffected



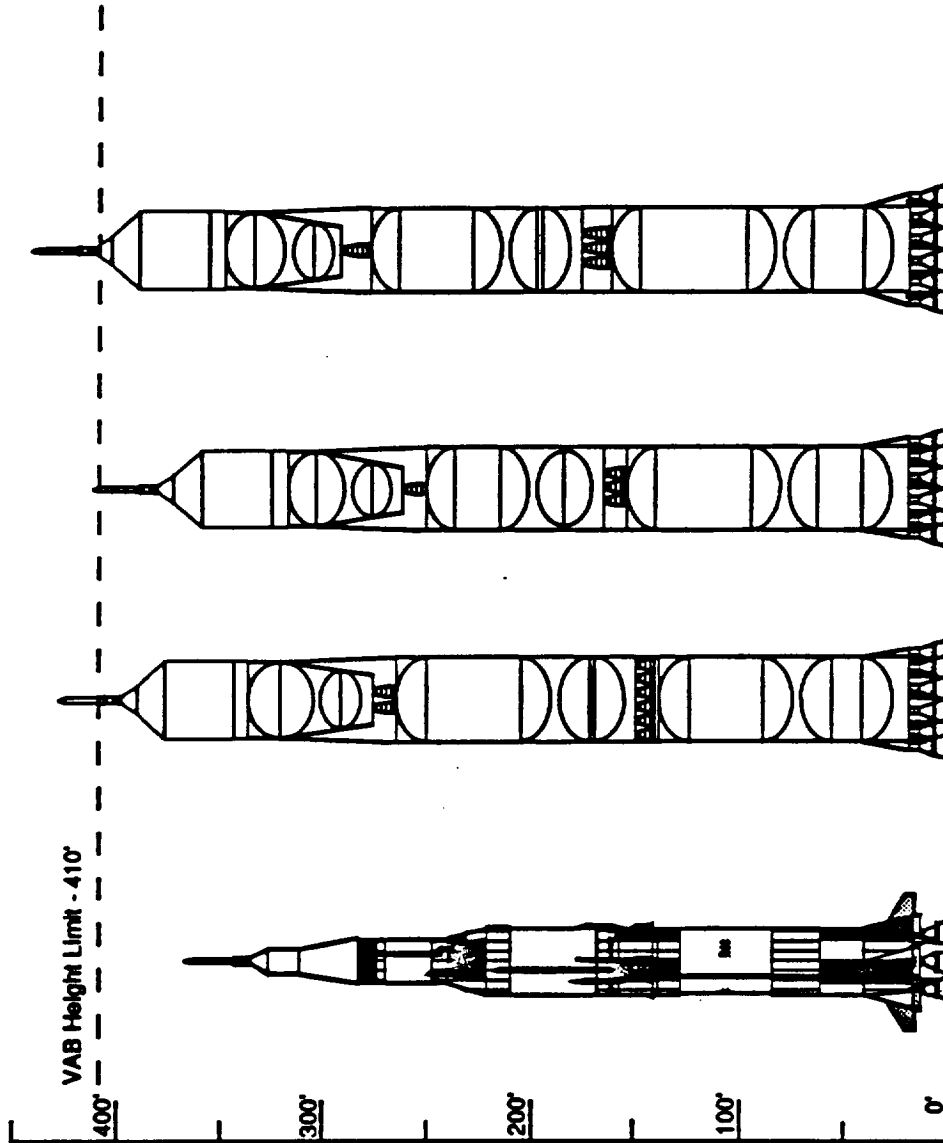
3-STAGE LUNAR HLLV OPTIONS Constant Diameter Stages



(Stage Dia. / Engine)					Post-TLJ P/L	47 t	94 t	100 t	101 t	96 t
Stage I	33.0 ft / 5 F-1	40.0 ft / 8 F-1A	43.0 ft / 9 F-1A	40.0 ft / 8 F-1A						
Stage II	33.0 ft / 5 J-2	40.0 ft / 13 J-2S	43.0 ft / 10 J-2S	40.0 ft / 5 SSME						
Stage III	21.7 ft / 1 J-2	40.0 ft / 2 J-2S	43.0 ft / 2 J-2S	40.0 ft / 1 SSME						



3-STAGE LUNAR HLLV OPTIONS Constant Diameter Stage I & II



(Stage Dia. / Engine)

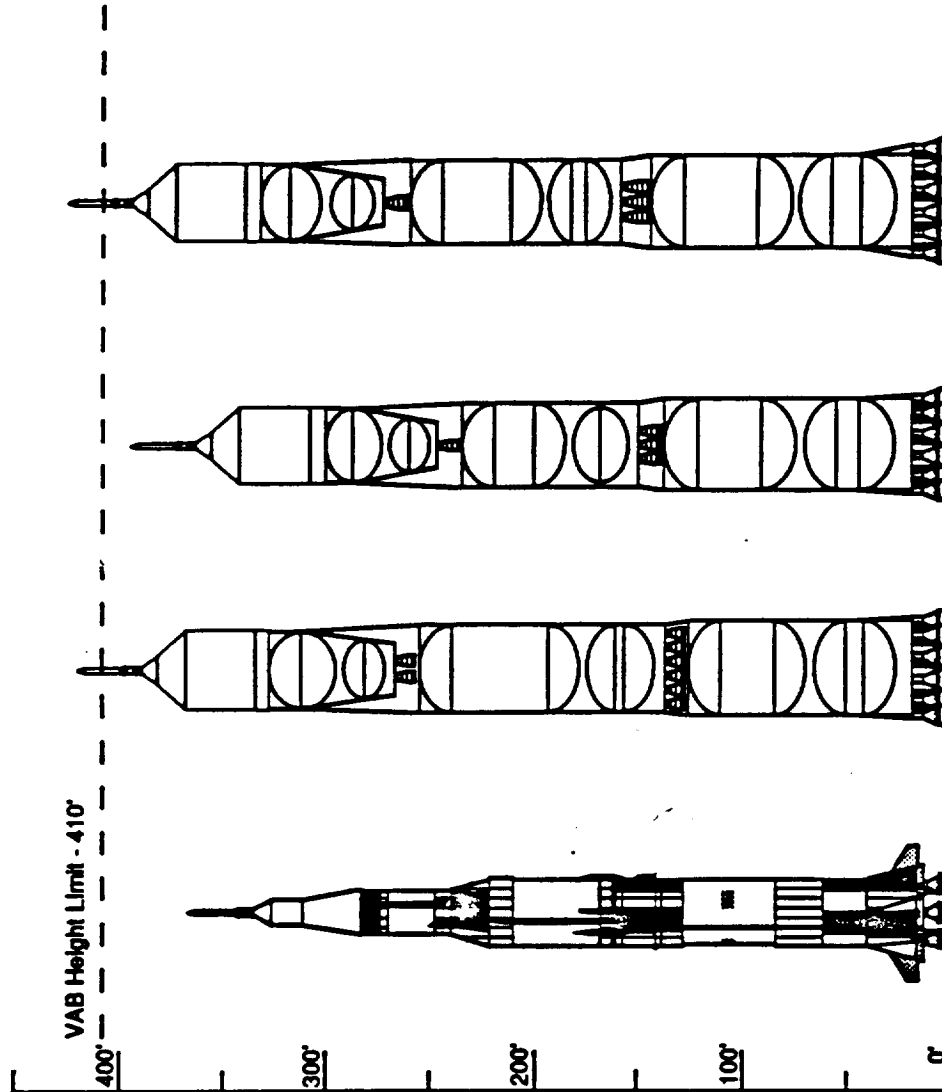
Stage I	33.0 ft / 5 F-1	42.0 ft / 8 F-1A	42.0 ft / 9 F-1A
Stage II	33.0 ft / 5 J-2	42.0 ft / 5 SSME	42.0 ft / 4 STME
Stage III	21.7 ft / 1 J-2	38.0 ft / 2 J-2S	38.0 ft / 1 STME

Post-TLJ P/L 471 931 1011 971

61

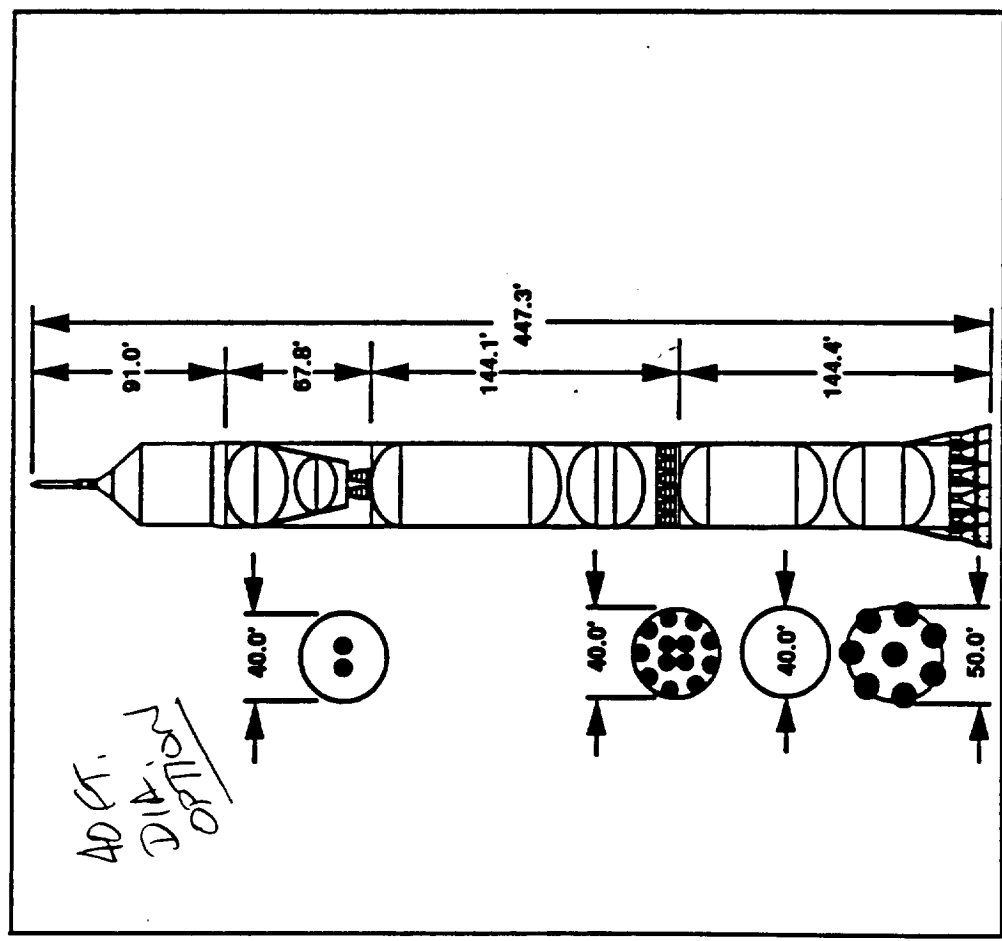


3-STAGE LUNAR HLLV OPTIONS Variable Diameter Stages



(Stage Dia. / Engines)			
Stage I	33.0 ft / 5 F-1	46.0 ft / 8 F-1A	46.0 ft / 9 F-1A
Stage II	33.0 ft / 5 J-2	42.0 ft / 5 SSME	42.0 ft / 4 STME
Stage III	21.7 ft / 1 J-2	38.0 ft / 2 J-2S	38.0 ft / 1 STME
Post-TLJ P/L	471	1001	961

3-Stage Lunar HLLV F-1A / J-2S Propulsion - Constant Diameter Stages



- NOTES:**
- AL 2219 Structure
 - No Engine Throttling Required
 - 72 deg. Launch Azimuth
 - No Engine-Out Capability

Payload:	206 Klb. (94 t) / 654 Klb. (297 t)
Final Position:	TLJ / 100 nm. Circ. Cutoff
GLOW:	12.0 Mib.
Max. G / Max. Q:	3.5 / 900 psf.

SHROUD
Diameter / Length:
Mass: 37.7 ft. / 47.4 ft.
23,860 lb.

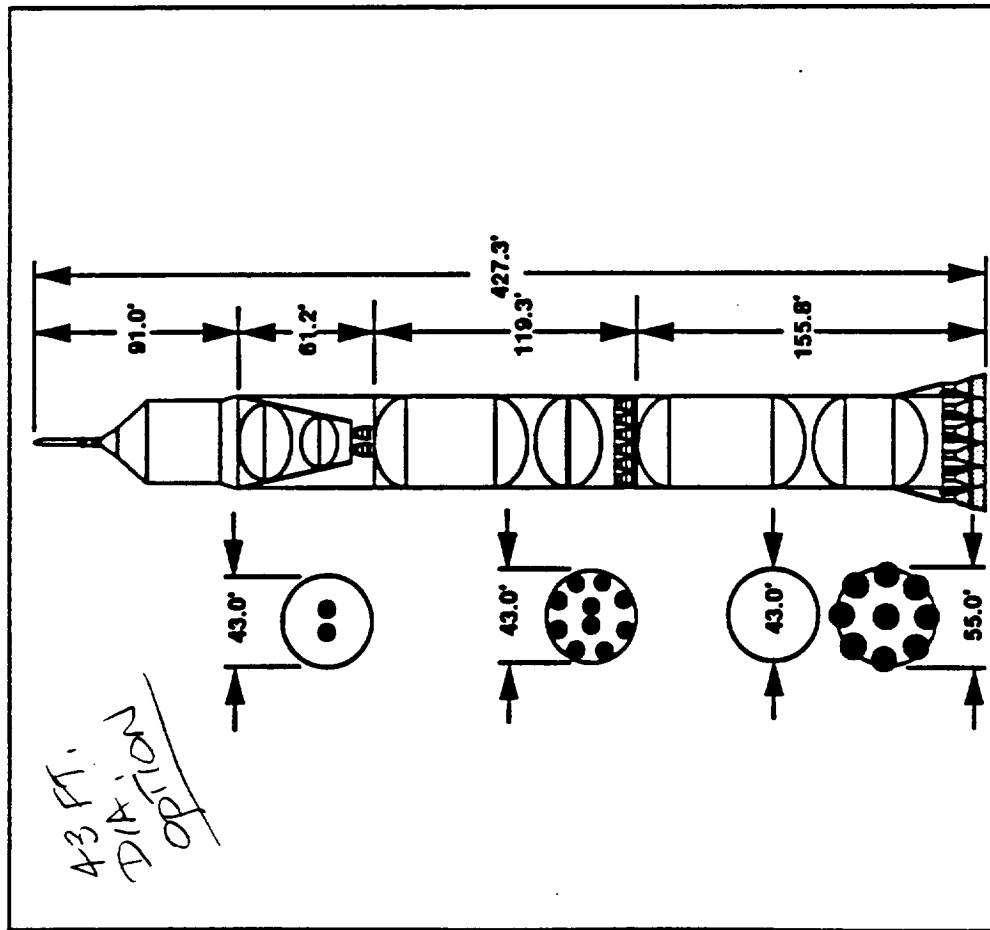
STAGE I:
Engine Type / #: F-1A / 8
Vac. / SL Thrust (ea): 2.02 / 1.8 Mib.
Propellant Type: RP-1 / LOX
Propellant Mass: 7.36 Mib.
Inert Mass: 579.0 Klb.
Interstage Mass: 13.0 Klb.
Reusability: None

STAGE II:
Engine Type / #: J-2S / 13
Vac. Thrust (ea): 265 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 2.75 Mib.
Inert Mass: 280.6 Klb.
Interstage Mass: 13.1 Klb.
Reusability: None

STAGE III:
Engine Type / #: J-2S / 2
Vac. Thrust (ea): 265 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 684.0 Klb.
Inert Mass: 92.4 Klb.
Reusability: None

3-Stage Lunar HLLV

F-1A / J-2S Propulsion - Constant Diameter Stages



NOTES:

- AL 2219 Structure
- F-1A Engines Throttled to 75% RPL for G-Control
- 72 deg. Launch Azimuth
- No Engine-Out Capability

Payload: 220 Klb. (100 t) / 652 Klb. (296 t)
Final Position: TLJ / 100 nm. Circ. Cutoff
GLOW: 12.0 Mib.
Max. G / Max. Q: 4.0 / 900 psf.

SHROUD
Diameter / Length: 37.7 ft. / 47.4 ft.
Mass: 23,860 lb.

STAGE I:

Engine Type / #: F-1A (9)
Vac. / SL Thrust (ea): 2.02 / 1.8 Mib.
Propellant Type: RP-1 / LOX
Propellant Mass: 9.39 Mib.
Inert Mass: 678.2 Klb.
Interstage Mass: 14.0 Klb.
Reusability: None

STAGE II:

Engine Type / #: J-2S (10)
Vac. Thrust (ea): 265 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 2.40 Mib.
Inert Mass: 245.8 Klb.
Interstage Mass: 14.0 Klb.
Reusability: None

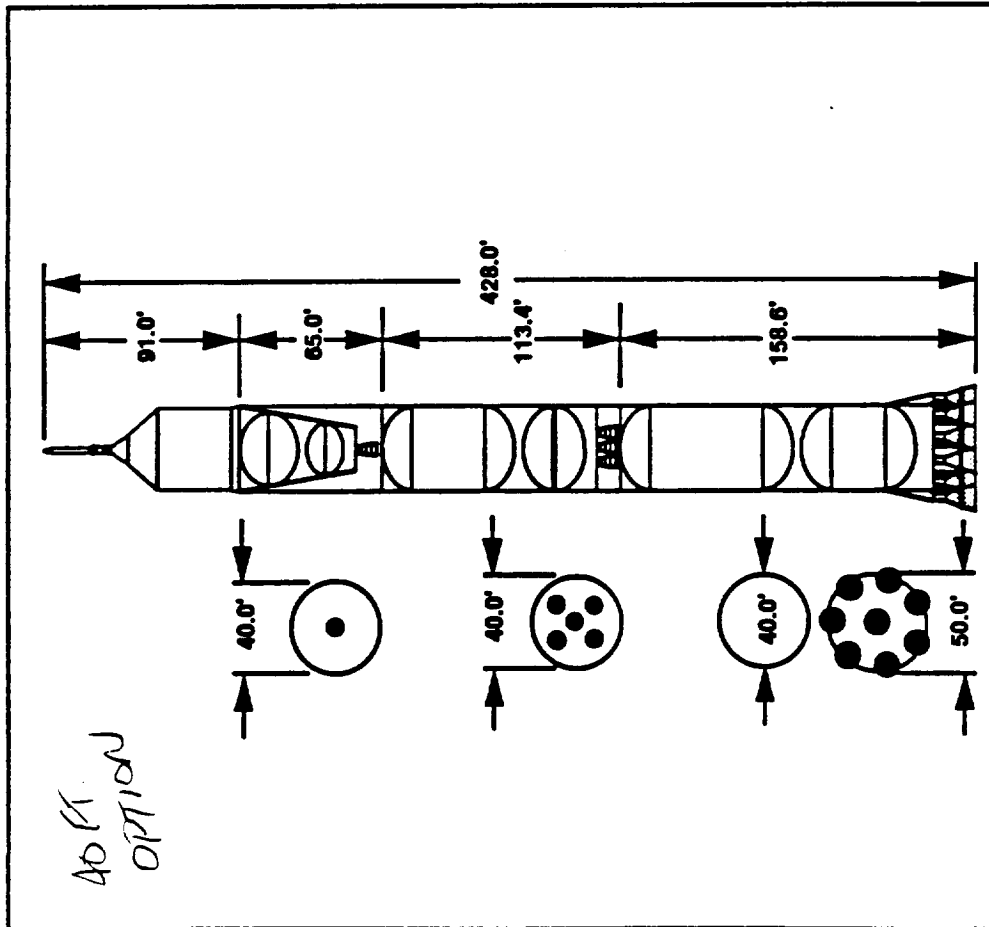
STAGE III:

Engine Type / #: J-2S (2)
Vac. Thrust (ea): 265 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 454.0 Klb.
Inert Mass: 78.5 Klb.
Reusability: None

3-Stage Lunar HLLV F-1A / SSME Propulsion - Constant Diameter Stages



NASA



- NOTES:**
- AL 2219 Structure
 - F-1A Engines Throttled to 75% RPL for G-Control
 - 72 deg. Launch Azimuth
 - No Engine-Out Capability

Payload: 222 Klb. (101 t) / 627 Klb. (284 t)
Final Position: TLI / 100 nm. Circ. Cutoff

GLOW: 12.0 Milb.
Max. G / Max. Q: 4.0 / 900 psf.

SHROUD
Diameter / Length: 37.7 ft. / 47.4 ft.
Mass: 23,860 lb.

STAGE I:
Engine Type / #: F-1A / 8
Vac. / SL Thrust (ea): 2.02 / 1.8 Milb.
Propellant Type: RP-1 / LOX
Propellant Mass: 8.46 Milb.
Inert Mass: 607.2 Klb.
Interstage Mass: 16.6 Klb.
Reusability: None

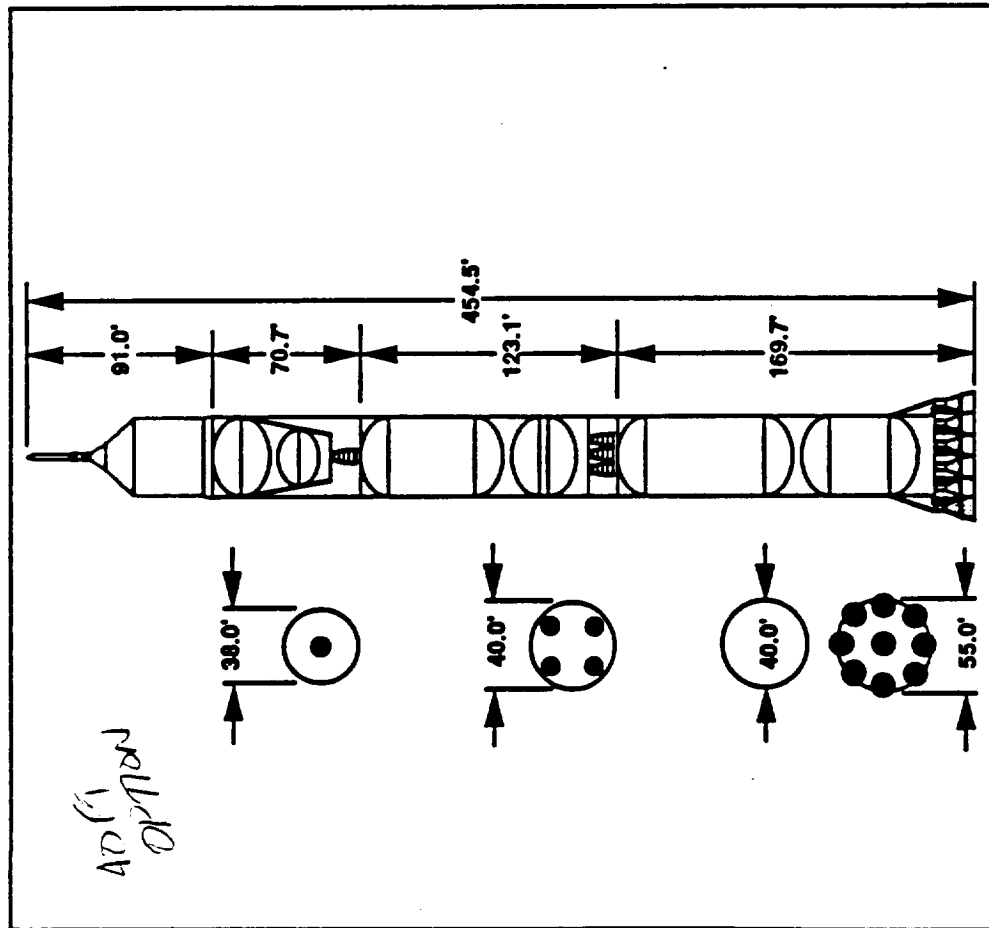
STAGE II:
Engine Type / #: SSME / 5
Vac. Thrust (ea): 470 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 1.95 Milb.
Inert Mass: 206.7 Klb.
Interstage Mass: 16.6 Klb.
Reusability: None

STAGE III:
Engine Type / #: SSME / 1
Vac. Thrust (ea): 470 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 432.3 Klb.
Inert Mass: 73.7 Klb.
Reusability: None



3-Stage Lunar HLLV

F-1A / STME Propulsion - Constant Diameter Stages



NOTES:

- AL 2219 Structure
- F-1A Engines Throttled to 75% RPL for G-Control
- 72 deg. Launch Azimuth
- No Engine-Out Capability

Payload: 211 Klb. (96 t) / 682 Klb. (309 t)
Final Position: TLJ / 100 nm. Circ. Cutoff

GLOW: 13.5 Mib.
Max. G / Max. Q: 4.0 / 900 psf.

SHROUD

Diameter / Length: 37.7 ft. / 47.4 ft.
Mass: 23,860 lb.

STAGE I:

Engine Type / #: F-1A / 9
Vac. / SL Thrust (ea): 2.02 / 1.8 Mib.
Propellant Type: RP-1 / LOX
Propellant Mass: 9.30 Mib.
Inert Mass: 672.3 Klb.
Interstage Mass: 15.8 Klb.
Reusability: None

STAGE II:

Engine Type / #: STME / 4
Vac. Thrust (ea): 650 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 2.24 Mib.
Inert Mass: 229.7 Klb.
Interstage Mass: 15.8 Klb.
Reusability: None

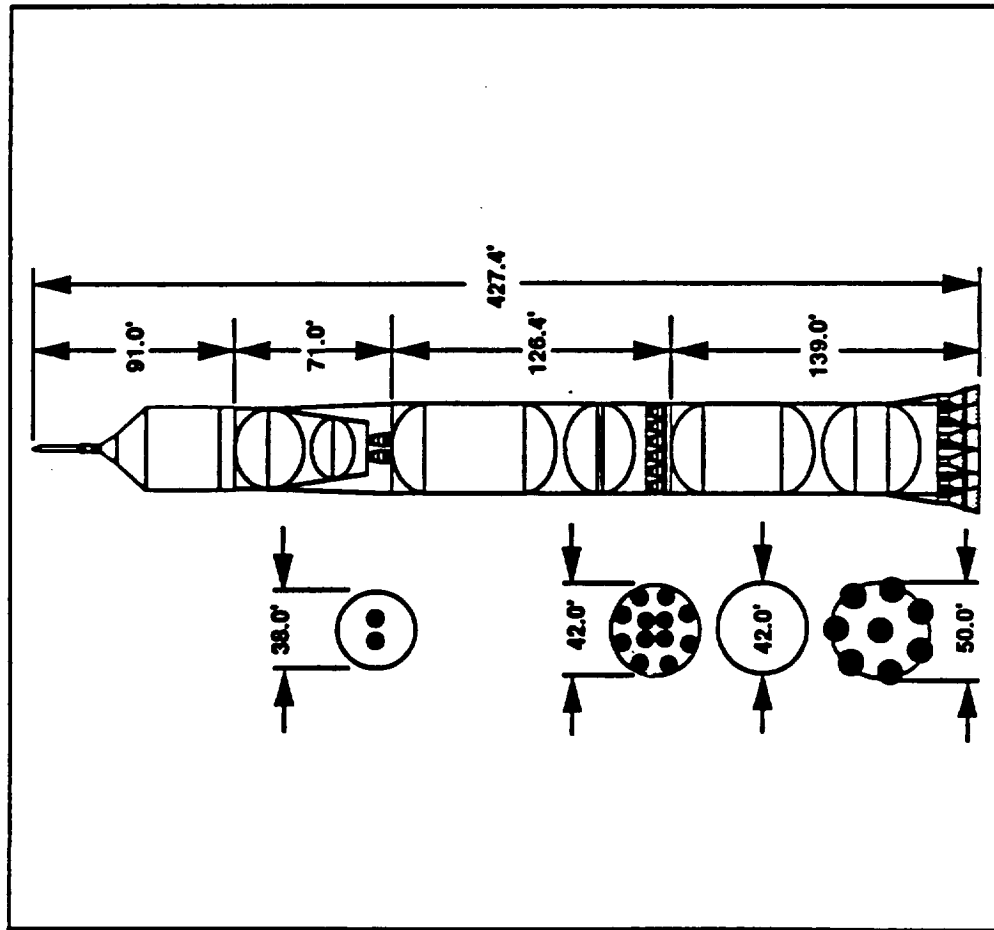
STAGE III:

Engine Type / #: STME / 1
Vac. Thrust (ea): 650 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 705.3 Klb.
Inert Mass: 96.5 Klb.
Reusability: None



3-Stage Lunar HLLV

F-1A / J-2S Propulsion - Constant Diameter Stage I & II



- NOTES:**
- AL 2219 Structure
 - No Engine Throttling Required
 - 72 deg. Launch Azimuth
 - No Engine-Out Capability

Payload:	205 Klb. (93 t) / 648 Klb. (294 t)
Final Position:	TLJ / 100 nm. Circ. Cutoff
GLOW:	12.0 Mib.
Max. G / Max. Q:	3.6 / 900 psf.

SHROUD
Diameter / Length:
Mass: 37.7 ft. / 47.4 ft.
23,860 lb.

STAGE I:
Engine Type / #: F-1A / 8
Vac. / SL Thrust (ea): 2.02 / 1.8 Mib.
Propellant Type: RP-1 / LOX
Propellant Mass: 7.58 Mib.
Inert Mass: 588.2 Klb.
Interstage Mass: 13.7 Klb.
Reusability: None

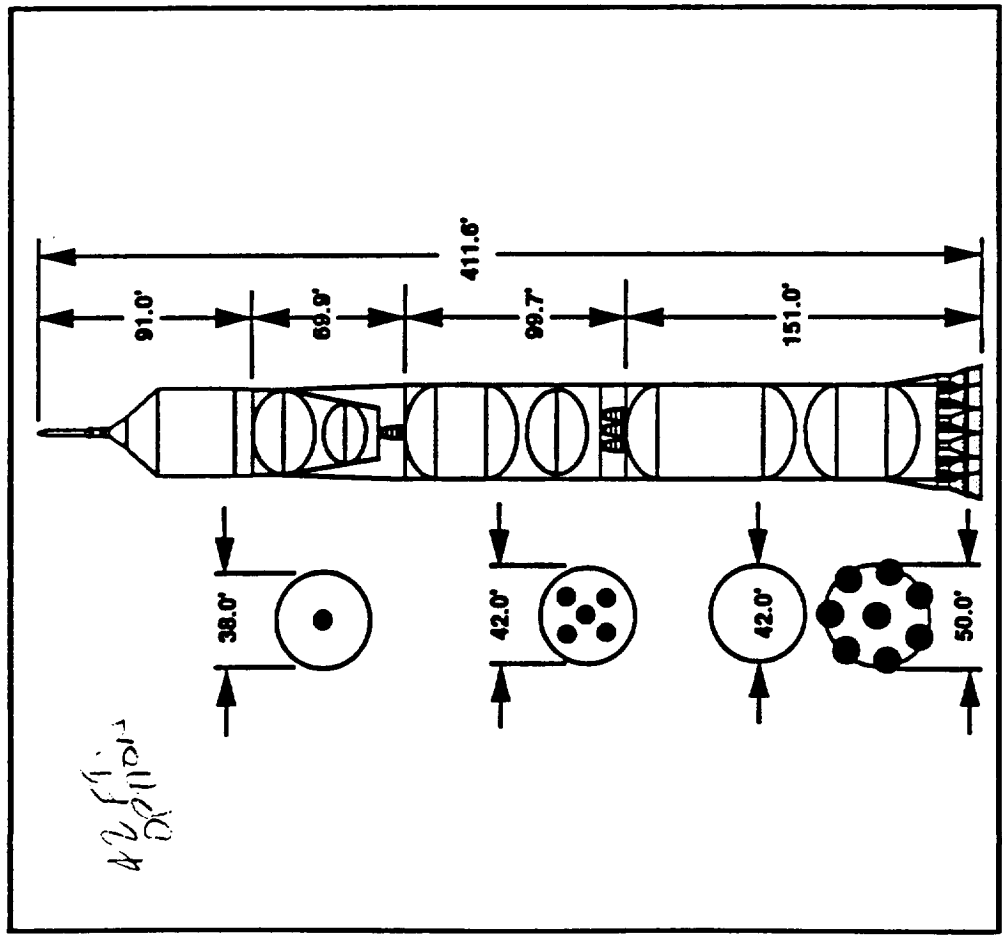
STAGE II:
Engine Type / #: J-2S / 12
Vac. Thrust (ea): 265 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 2.51 Mib.
Inert Mass: 264.0 Klb.
Interstage Mass: 13.1 Klb.
Reusability: None

STAGE III:
Engine Type / #: J-2S / 2
Vac. Thrust (ea): 265 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 718.9 Klb.
Inert Mass: 91.7 Klb.
Reusability: None



NASA F-1A / SSME Propulsion - Constant Diameter Stage I & II

3-Stage Lunar HLLV



Payload:	222 Kib. (101 t) / 650 Kib. (295 t)
Final Position:	TLJ / 100 nm. Circ. Cutoff
GLOW:	12.0 Mib.
Max. G / Max. Q:	4.0 / 900 psf.

SHROUD
Diameter / Length:
Mass: 37.7 ft. / 47.4 ft.
23,860 lb.

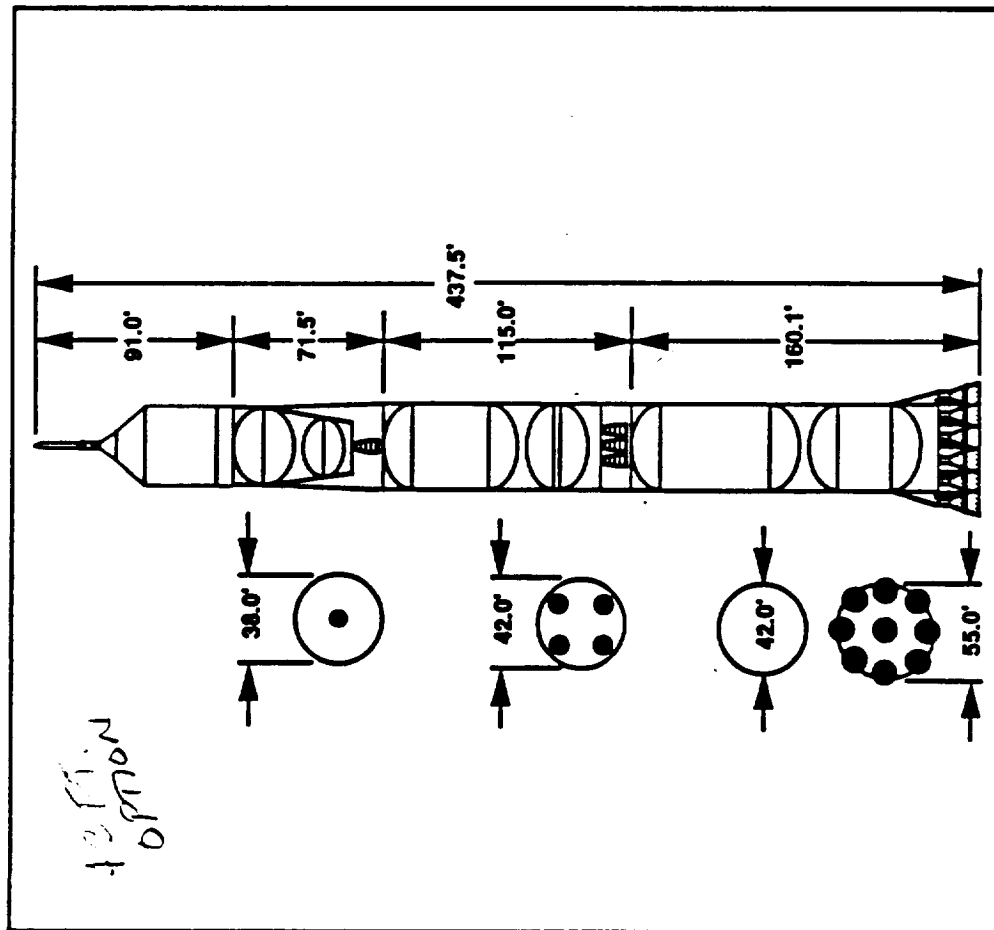
STAGE I:
Engine Type / #: F-1A / 8
Vac. / SL Thrust (ea): 2.02 / 1.8 Mib.
Propellant Type: RP-1 / LOX
Propellant Mass: 8.59 Mib.
Inert Mass: 614.2 Kib.
Interstage Mass: 17.5 Kib.
Reusability: None

STAGE II:
Engine Type / #: SSME / 5
Vac. Thrust (ea): 470 Kib.
Propellant Type: LH2 / LOX
Propellant Mass: 1.64 Mib.
Inert Mass: 194.1 Kib.
Interstage Mass: 16.7 Kib.
Reusability: None

STAGE III:
Engine Type / #: SSME / 1
Vac. Thrust (ea): 470 Kib.
Propellant Type: LH2 / LOX
Propellant Mass: 620.1 Kib.
Inert Mass: 83.7 Kib.
Reusability: None

- NOTES:**
- AL 2219 Structure
 - F-1A Engines Throttled to 75% RPL for G-Control
 - 72 deg. Launch Azimuth
 - No Engine-Out Capability

F-1A / STME Propulsion - Constant Diameter Stage I & II



- NOTES:**
- AL 2219 Structure
 - F-1A Engines Throttled to 75% RPL for G-Control
 - 72 deg. Launch Azimuth
 - No Engine-Out Capability

214 Klb. (97 t) / 679 Klb. (308 t)	13.5 Mlb.
TLJ / 100 nm. Circ. Cutoff	4.0 / 900 psf.
Final Position:	
GLOW:	
Max. G / Max. Q:	

SHROUD
Diameter / Length:
Mass:

**37.7 ft. / 47.4 ft.
23,860 lb.**

STAGE1:
Engine Type / #:
Vac. / SL Thrust (ea):
Propellant Type:
Propellant Mass:
Inert Mass:
Interstage Mass:
Reusability:

F-1A / 9
2.02 / 1.8 Mlb.
RP-1 / LOX
9.36 Mlb.
677.3 Klb.
16.6 Klb.
None

STAGE II:
Engine Type / #:
Vac. Thrust (ea):
Propellant Type:
Propellant Mass:
Inert Mass:
Interstage Mass:
Reusability:

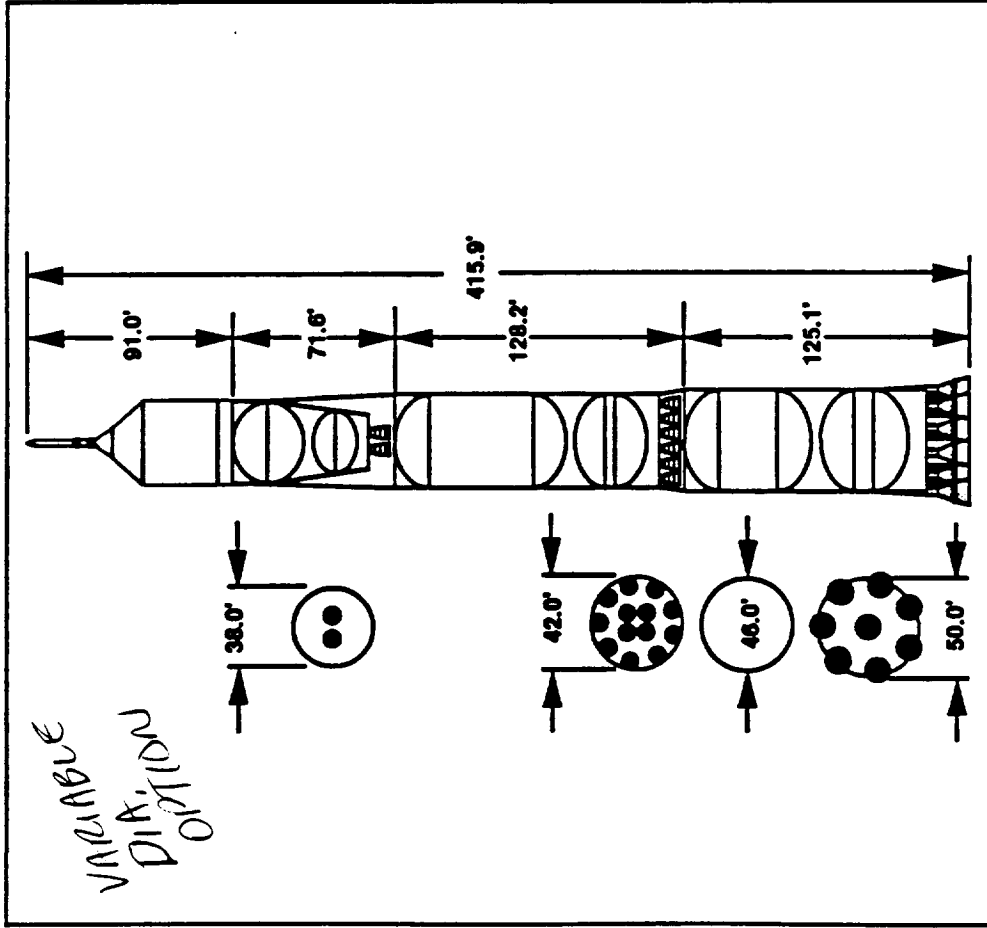
STME / 4
650 Kib.
LH2 / LOX
2.20 Mib.
231.2 Kib.
15.9 Kib.
None

STAGE III:
Engine Type / #:
Vac. Thrust (ea):
Propellant Type:
Propellant Mass:
Inert Mass:
Reusability:

STME / 1
650 Kib.
LH2 / LOX
685.1 Kib.
92.7 Kib.
None

3-Stage Lunar HLLV

F-1A / J-2S Propulsion - Variable Diameter Stages



- NOTES:**
- AL 2219 Structure
 - No Engine Throttling Required
 - 72 deg. Launch Azimuth
 - No Engine-Out Capability

Payload: 208 Klb. (94 t) / 658 Klb. (298 t)
Final Position: TLI / 100 nm. Circ. Cutoff
GLOW: 12.0 Mlb.
Max. G / Max. Q: 3.6 / 900 psf.

SHROUD
Diameter / Length: 37.7 ft. / 47.4 ft.
Mass: 23,860 lb.

STAGE I:
Engine Type / #: F-1A / 8
Vac. / SL Thrust (ea): 2.02 / 1.8 Mlb.
Propellant Type: RP-1 / LOX
Propellant Mass: 7.50 Mlb.
Inert Mass: 593.8 Klb.
Interstage Mass: 14.4 Klb.
Reusability: None

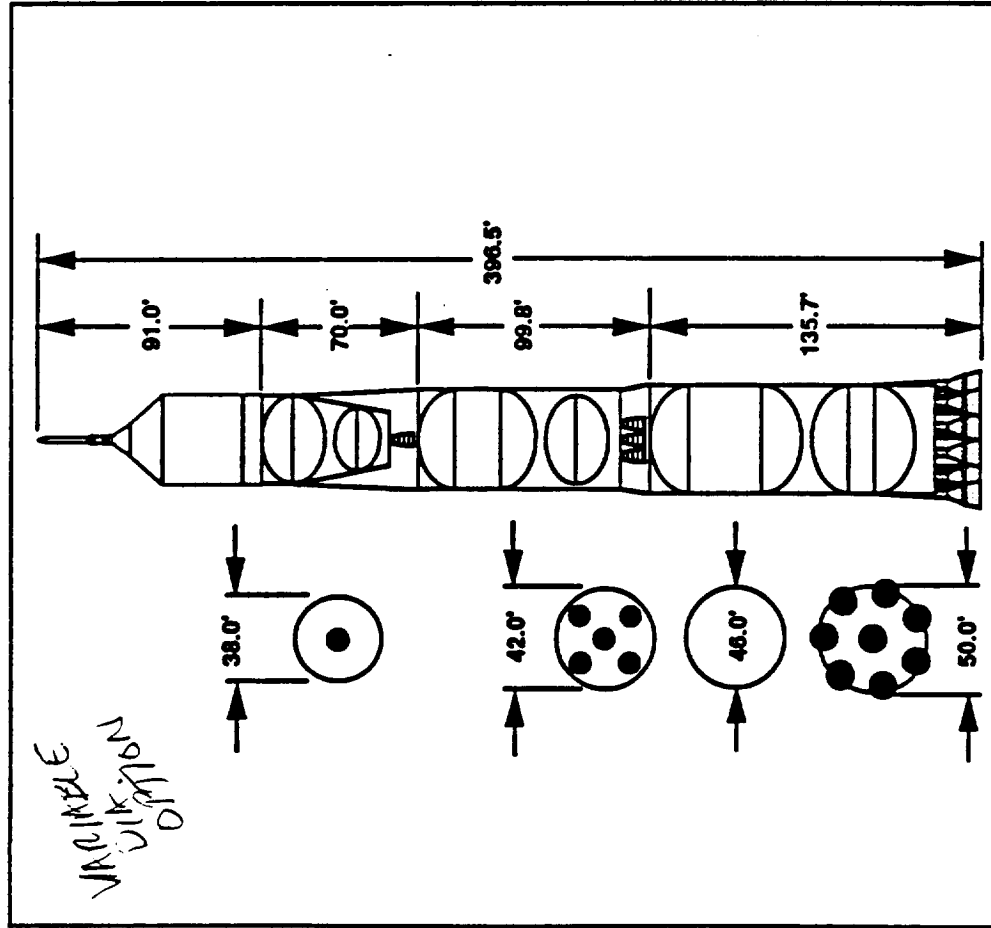
STAGE II:
Engine Type / #: J-2S / 13
Vac. Thrust (ea): 265 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 2.56 Mlb.
Inert Mass: 273.8 Klb.
Interstage Mass: 13.1 Klb.
Reusability: None

STAGE III:
Engine Type / #: J-2S / 2
Vac. Thrust (ea): 265 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 734.3 Klb.
Inert Mass: 92.6 Klb.
Reusability: None



3-Stage Lunar HLLV

F-1A / SSME Propulsion - Variable Diameter Stages



NOTES:

- AL 2219 Structure
- F-1A Engines Throttled to 75% RPL for G-Control
- 72 deg. Launch Azimuth
- No Engine-Out Capability

Payload: 221 Kib. (100 t) / 647 Kib. (294 t)
Final Position: TLI / 100 nm. Circ. Cutoff

GLOW: 12.0 Mib.
Max. G / Max. Q: 4.0 / 900 psf.

SHROUD
Diameter / Length: 37.7 ft. / 47.4 ft.
Mass: 23,860 lb.

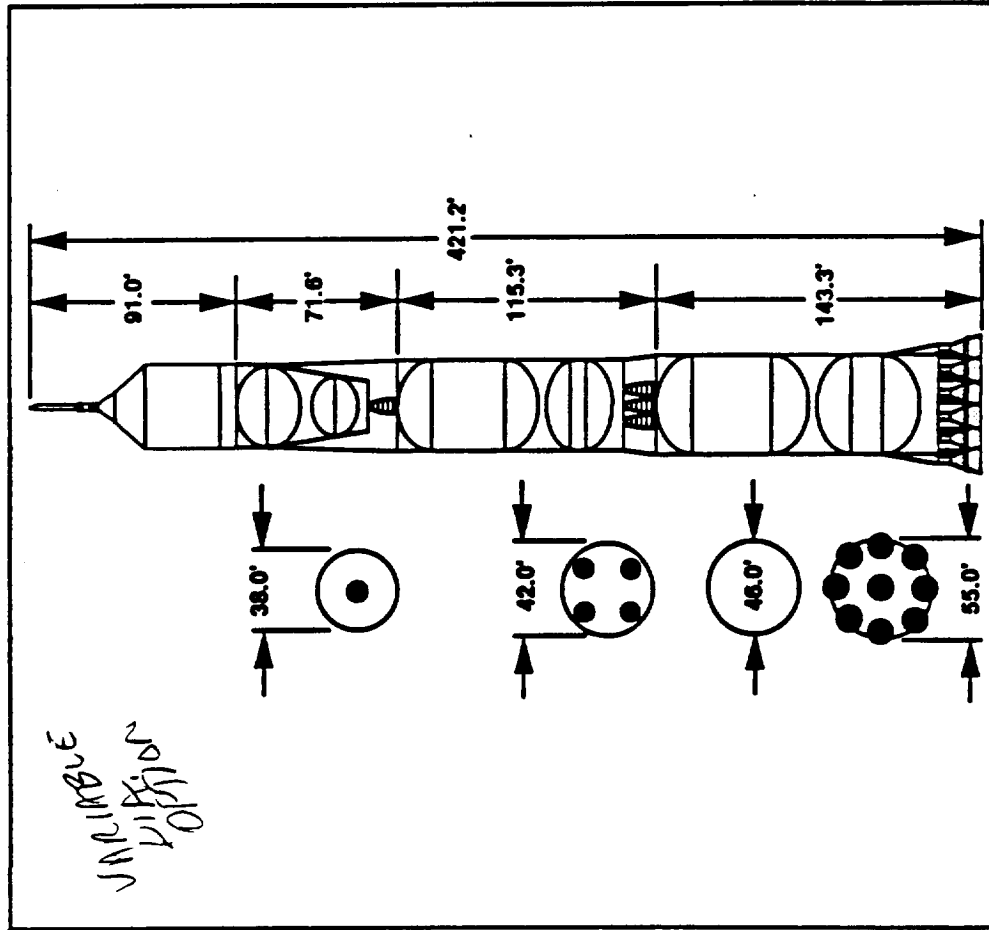
STAGE I:
Engine Type / #: F-1A / 8
Vac. / SL Thrust (ea): 2.02 / 1.8 Mib.
Propellant Type: RP-1 / LOX
Propellant Mass: 8.57 Mib.
Inert Mass: 621.5 Kib.
Interstage Mass: 18.4 Kib.
Reusability: None

STAGE II:
Engine Type / #: SSME / 5
Vac. Thrust (ea): 470 Kib.
Propellant Type: LH2 / LOX
Propellant Mass: 1.64 Mib.
Inert Mass: 194.5 Kib.
Interstage Mass: 16.7 Kib.
Reusability: None

STAGE III:
Engine Type / #: SSME / 1
Vac. Thrust (ea): 470 Kib.
Propellant Type: LH2 / LOX
Propellant Mass: 624.0 Kib.
Inert Mass: 83.9 Kib.
Reusability: None

3-Stage Lunar HLLV

F-1A / STME Propulsion - Variable Diameter Stages



- NOTES:**
- AL 2219 Structure
 - F-1A Engines Throttled to 75% RPL for G-Control
 - 72 deg. Launch Azimuth
 - No Engine-Out Capability

Payload:	212 Klb. (96 l) / 675 Klb. (306 l)
Final Position:	TLI / 100 nm. Circ. Cutoff
GLOW:	13.5 Mib.
Max. G / Max. Q:	4.0 / 900 psf.

SHROUD
Diameter / Length:
Mass: 37.7 ft. / 47.4 ft.
 23,860 lb.

STAGE I:
Engine Type / #: F-1A / 9
Vac. / SL Thrust (ea): 2.02 / 1.8 Mib.
Propellant Type: RP-1 / LOX
Propellant Mass: 9.34 Mib.
Inert Mass: 684.4 Klb.
Interstage Mass: 17.5 Klb.
Reusability: None

STAGE II:
Engine Type / #: STME / 4
Vac. Thrust (ea): 650 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 2.20 Mib.
Inert Mass: 231.6 Klb.
Interstage Mass: 15.9 Klb.
Reusability: None

STAGE III:
Engine Type / #: STME / 1
Vac. Thrust (ea): 650 Klb.
Propellant Type: LH2 / LOX
Propellant Mass: 689.2 Klb.
Inert Mass: 92.9 Klb.
Reusability: None



Introduction		SYSTEMS ENGINEERING DIVISION
		Edward A. Robertson
		July 1992

The design of the First Lunar Outpost (FLO) payload fairing (PLF) is influenced by:

- 1) Piloted and cargo vehicle configurations
 - 2) Packaging of the piloted and cargo vehicles and logistics within the PLF
 - 3) Crew escape operations
 - 4) HLLV/PLF aerodynamic performance and PLF separation
-



Vehicle Configuration		SYSTEMS ENGINEERING DIVISION
		Edward A. Robertson
		July 1992

The configuration of the FLO Piloted Vehicle is driven by flight operations:

- 1) Return stage mounted on "top" of the lander stage with a relatively clean mechanical interface
 - 2) Logistics carriers located at the perimeter of the vehicle to simplify lunar surface payload transfer
 - 3) Low center of gravity at touchdown promotes landing stability and a smaller landing gear tread radius
-



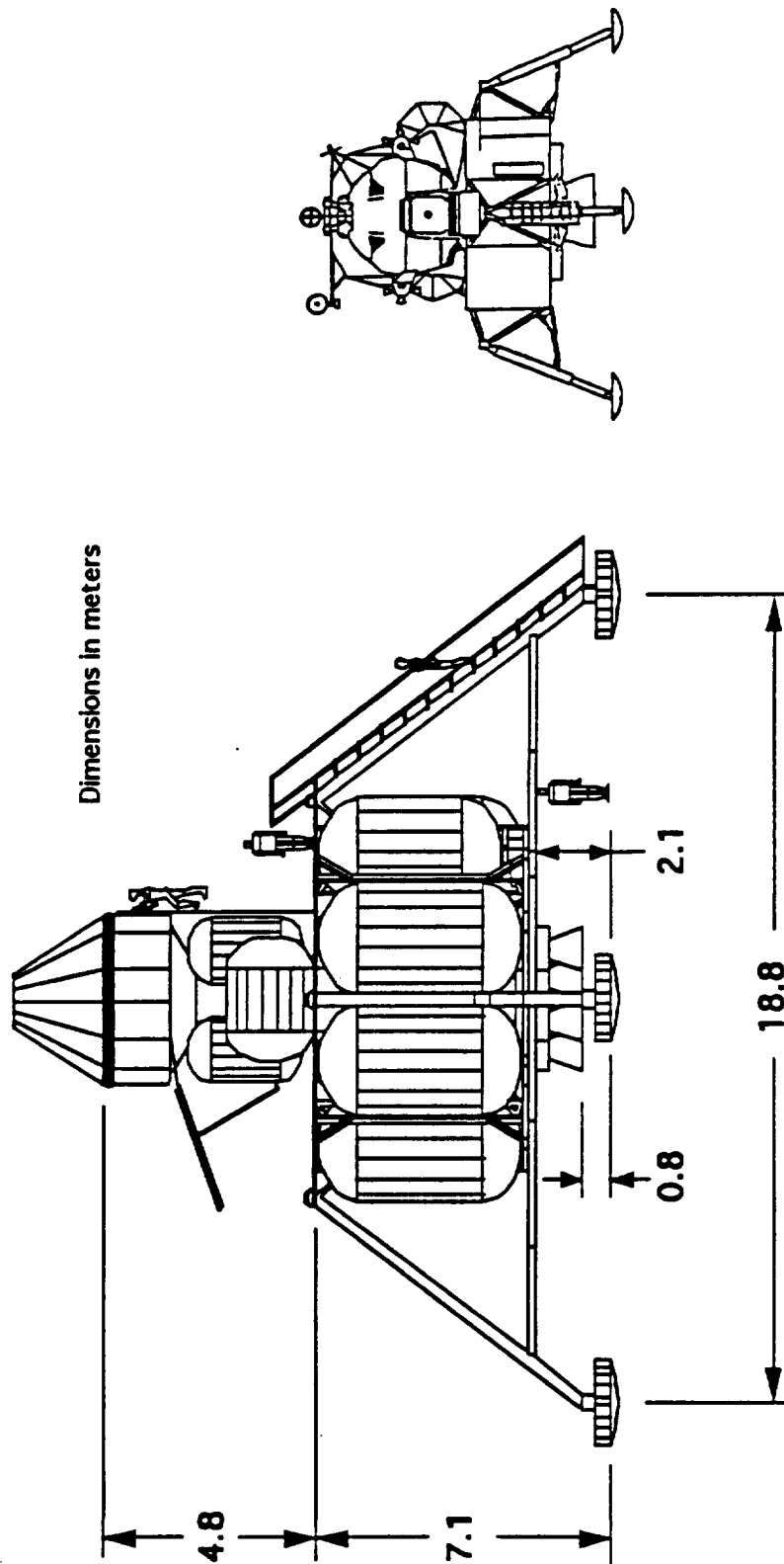
Johnson Space Center - Houston, Texas

SYSTEMS ENGINEERING DIVISION

Edward A. Robertson

July 1992

Vehicle Configuration

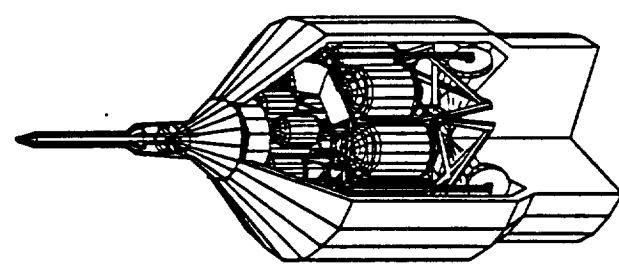


Scale comparison of FLO Piloted Vehicle and the Apollo LM



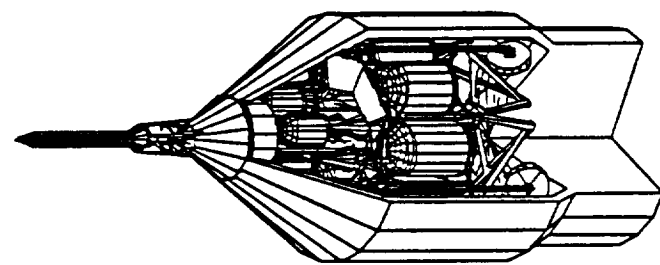
Johnson Space Center - Houston, Texas

Aerodynamics	
SYSTEMS ENGINEERING DIVISION	
Edward A. Robertson	July 1992



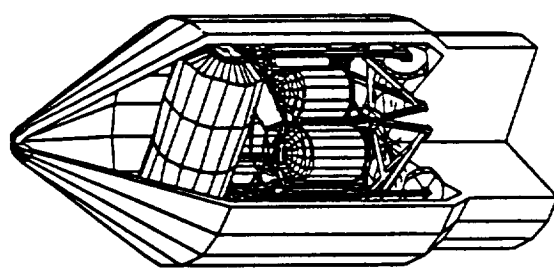
Original

(42 degree cone)
CM & LES Exposed
Piloted

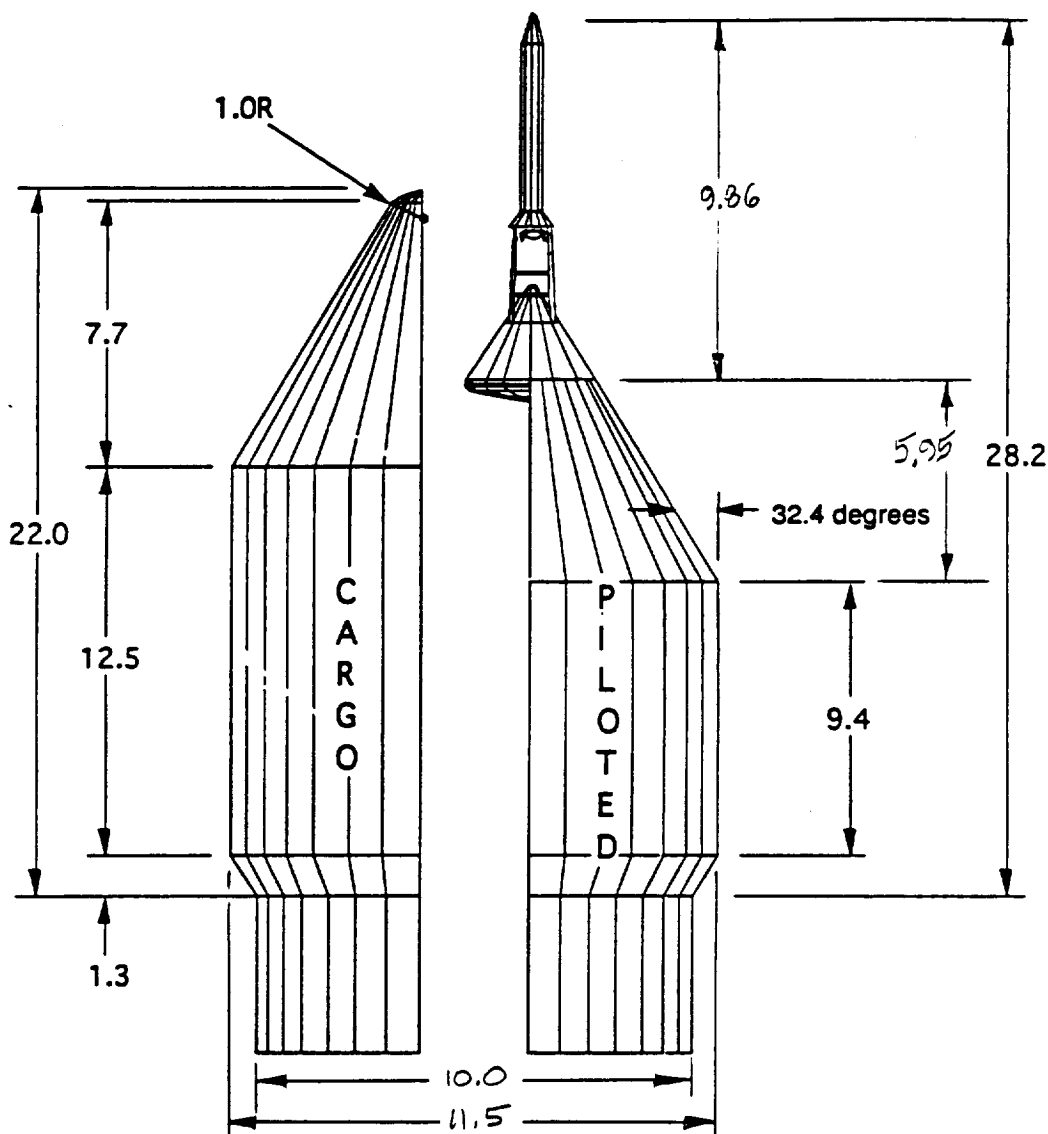


Modified*

(32.4 degree cone)
CM & LES Exposed
Piloted and Cargo



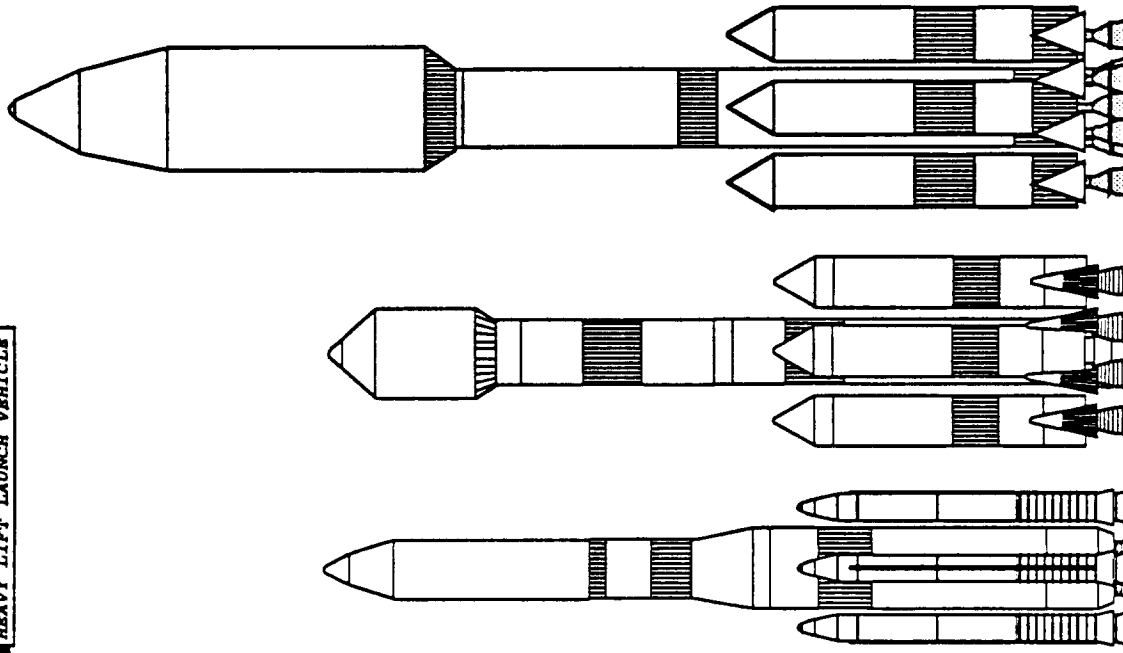
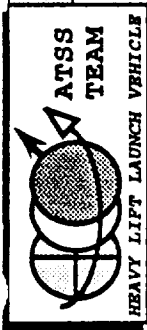
Aerodynamics



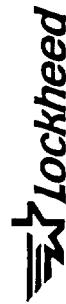
Modified Option #3: Exposed CM & LES
Cargo and Piloted PLFs

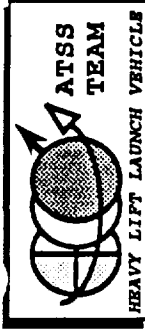
All linear dimensions in meters





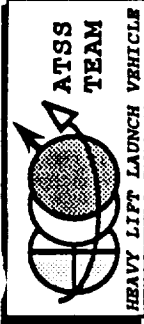
NASA/HEADQUARTERS PERCEPTIONS & DESIREMENTS





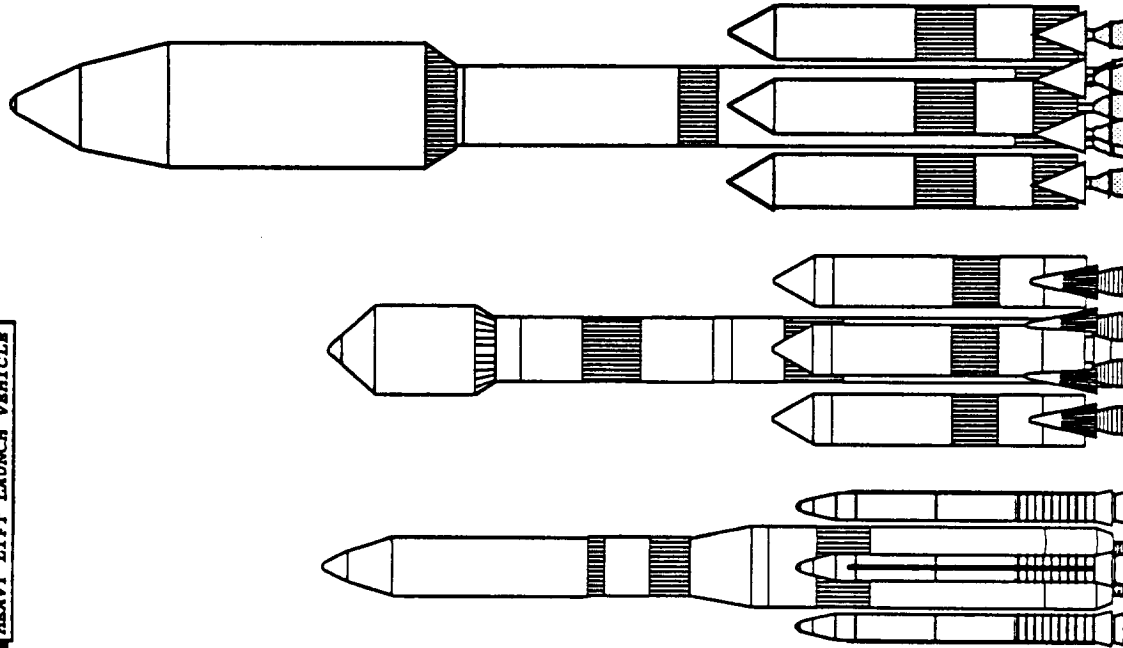
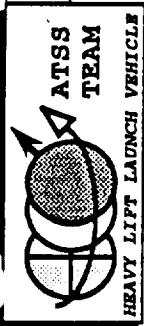
NASA/HEADQUARTER PERCEPTIONS & DESIREMENTS

- D. Goldin giving mixed signals on SEI support
 - Worked with M. Griffin on SDI
 - Supports basic premise of return to Moon and explore Mars
 - Is a big SSF supporter; wants to phase Space Shuttle out early (early 2000s; as opposed to 2020)
 - Supports mix of ELVs (NLS), PLS, & TBD cargo return vehicle as Shuttle replacement
 - Quoted as saying that return to Moon and Mars missions are "decades away"
 - Has told Griffin to utilize VAB for lunar and try to use it for Mars; implying minimum DDT&E philosophy
- Charter of NASA Red/Blue teams to reduce NASA program costs by 30% (with same number of programs) & Griffin heading up "Space Program Integration" Blue team has been read to mean that 30% savings will become Griffin's budget for SEI
- Griffin wants to test White House backing of SEI with aggressive "faster quicker, cheaper" approach by identifying requirements to go back to Moon to stay by early 2000s
 - If need to change/waive laws, so be it
 - Minimize new technologies; only use if cost-effective



NASA/HEADQUARTER PERCEPTIONS & DESIREMENTS (Concluded)

- Griffin is convinced that single-launch is the only cost-effective/minimum risk way to return to Moon
 - Does not accept "Cannot build/operate vehicles as large as is required for single-launch
 - Austin is a "convert" to this religion
- Griffin does not like solid propulsion; has no other propulsion biases
 - If CLS hardware is better, so be it
- The NASA FLO element teams are trying to take a "make it work" attitude
 - "Over 40" people have doubts
 - "Under 40" people say "can do"
 - JSC/MSFC rivalry hampers team play
 - KSC/SSC will take JSC/MSFC requirements and make them work
- Perception of FLO as a NASA in-house activity
 - Lockheed was one of first to penetrate (LSOC,MSD/LESC)
 - Martin STV contract changed over to TLI stage design contract
 - Lockheed participation at HLLV Subteam TIMs is touchy, but workable



TA-2 Overview



Advanced Transportation System Studies Heavy Lift Launch Vehicle Concept Development

Contract Kick-Off Briefing

**Technical Area - 2
NAS8-39208**

**Marshall Space Flight Center
Huntsville, Alabama
17 June 1992**



AGENDA

1. *Introduction*

Johnson/Trudeau
McCurry

2. *Study Overview*

- Description
- Organization
- Major Tasks
- Schedule

3. *Study Methodology*

- Vehicle Design and Performance Assessments
- Analysis Tool Benchmarks
- Ground Operations Assessments
- Parametric Cost Analysis
- Subsystem Definition and Technologies

Berning

Sagis

Letchworth

Skratt

Rustenburg

4. *FLO Assessment Results To-Date*

- NLS-Derived Common Diameter Concepts
- Monolithic Concepts
- Ground Operations Assessments

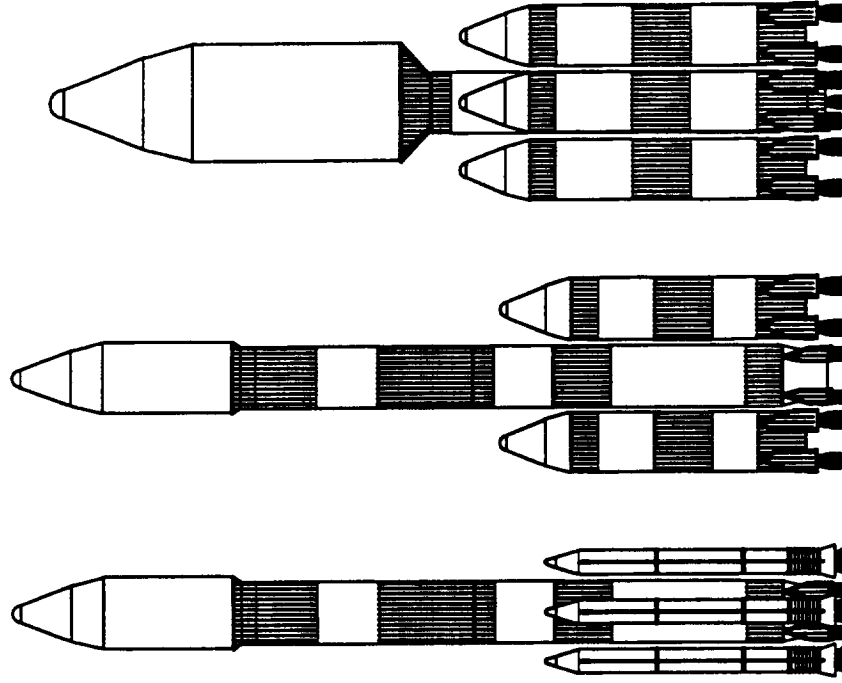
McCurry

Holden

Letchworth

5. *Concluding Remarks*

Johnson/Trudeau/McCurry

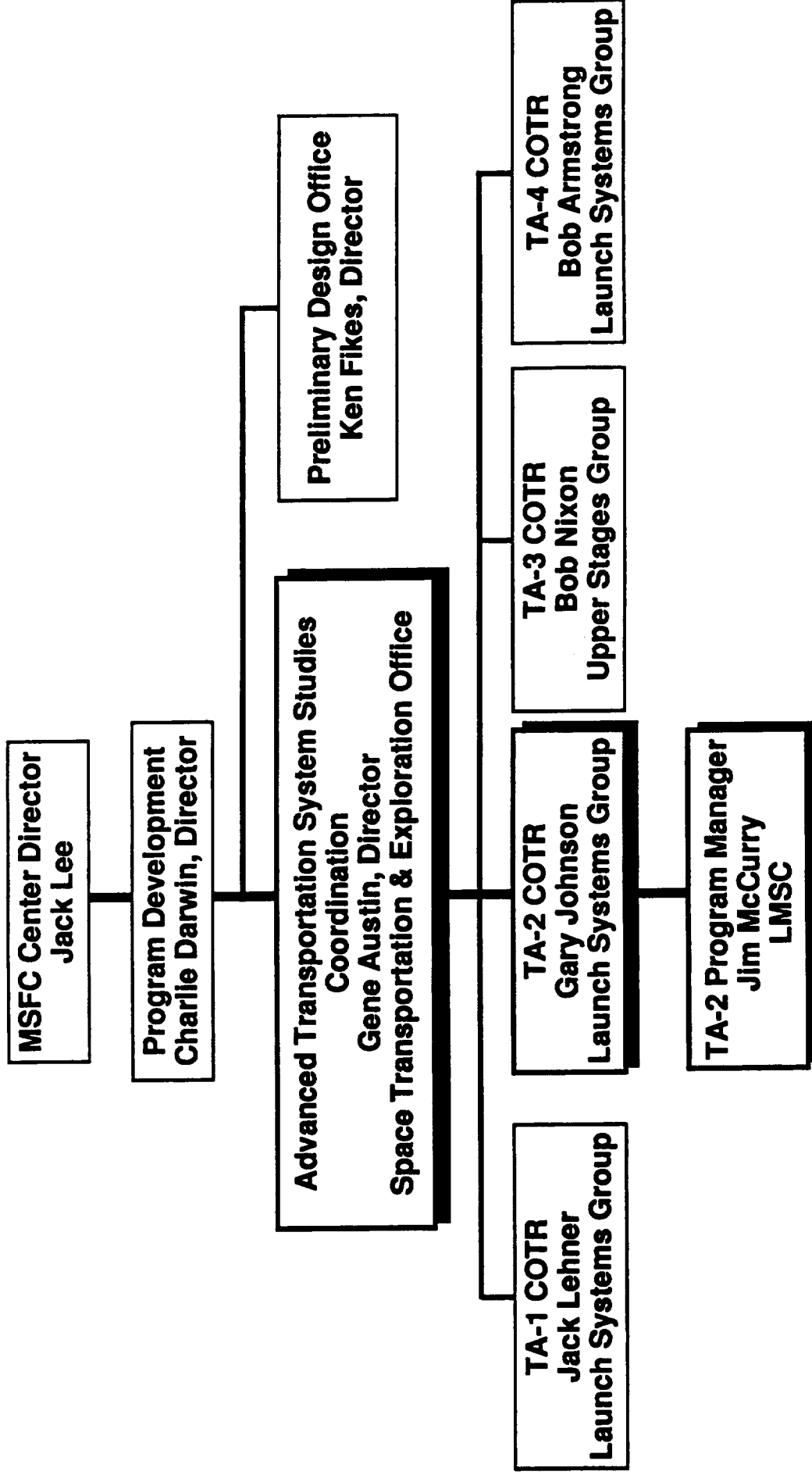


1. Introduction

G. Johnson/MSFC
H. Trudeau/Lockheed

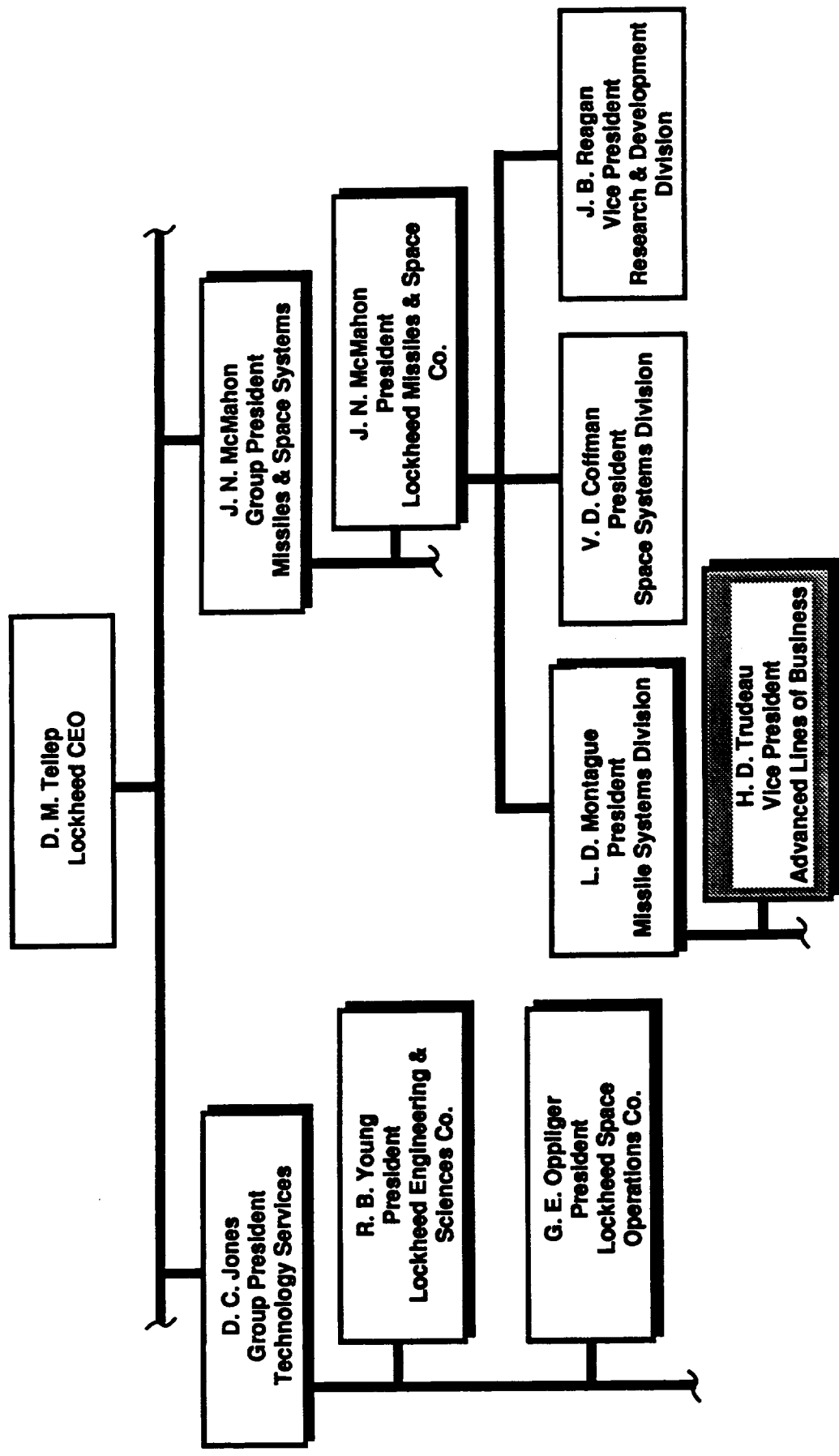


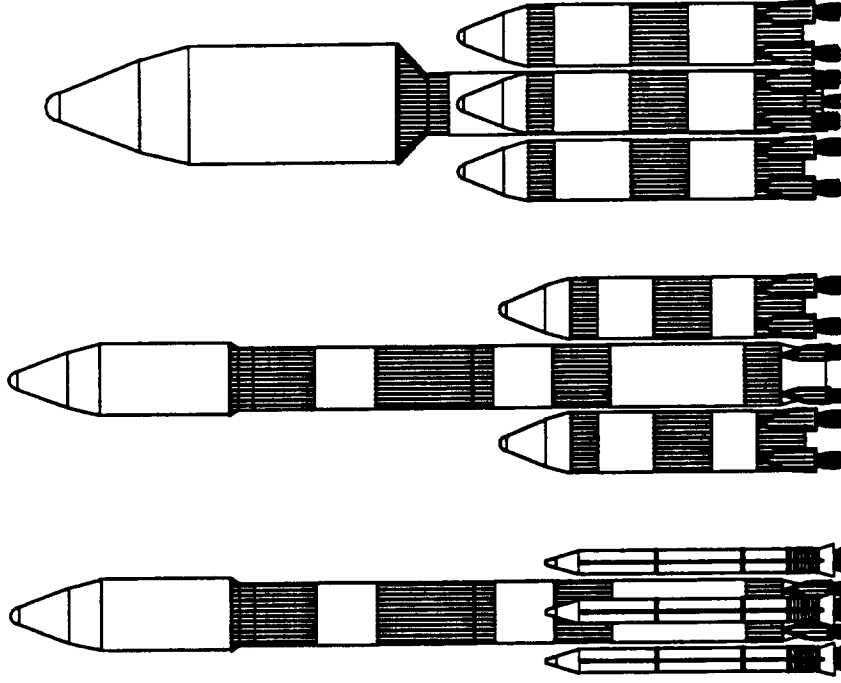
TA-2 Responsibility Flow-Down





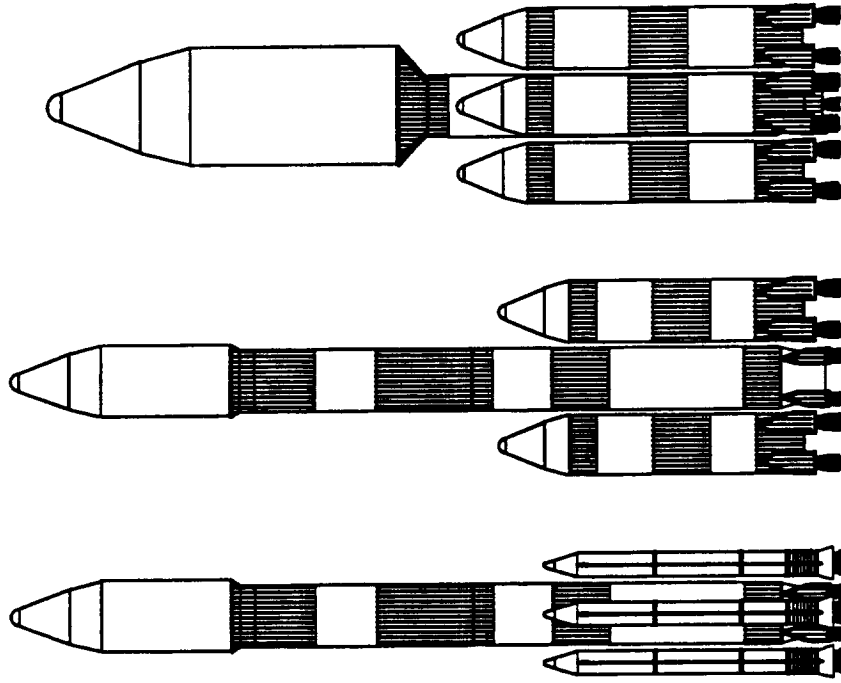
LOCKHEED CORPORATION





2. Study Overview

J. McCurry/Lockheed

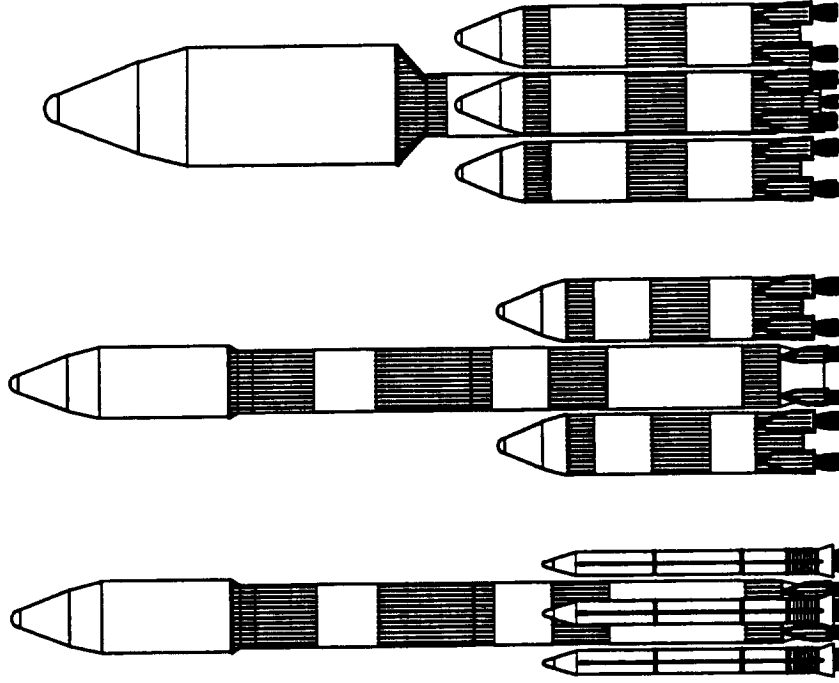


Study Description



Study Description

- Conceptual definition and assessment of candidate Heavy Lift Launch Vehicle (HLLV) configurations that meet the Level II requirements of the lunar and Mars Space Exploration Initiative (SEI)
 - Approximate payload range (200K-600K lbm; 90-250 t)
- Vehicle concepts to be analyzed include (at MSFC's discretion) NLS-derived, Saturn V-derived, clean-sheet, and foreign options
- Top-level vehicle and facility requirements, design, and cost sensitivities are to be identified
- Evolutionary paths to and from the preferred concepts are to be identified
- Technology and advanced development requirements are to be identified for the recommended vehicle concepts
- A wide range of concepts are to be assessed in the first year, with down-selection and further design definition in the subsequent two years
- Funding: First Year \$926K; Second Year \$949K; Third Year \$934K

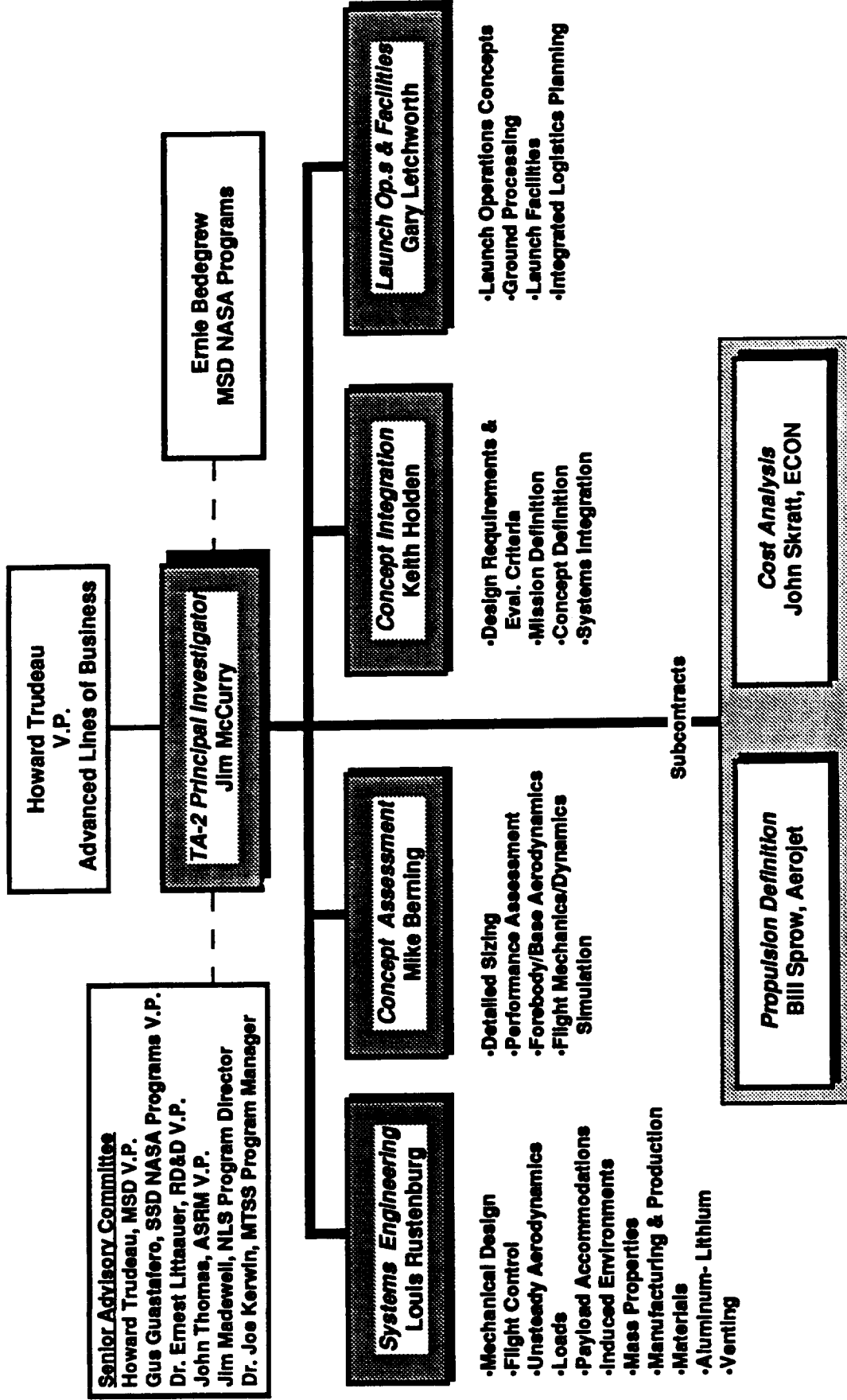


Study Organization



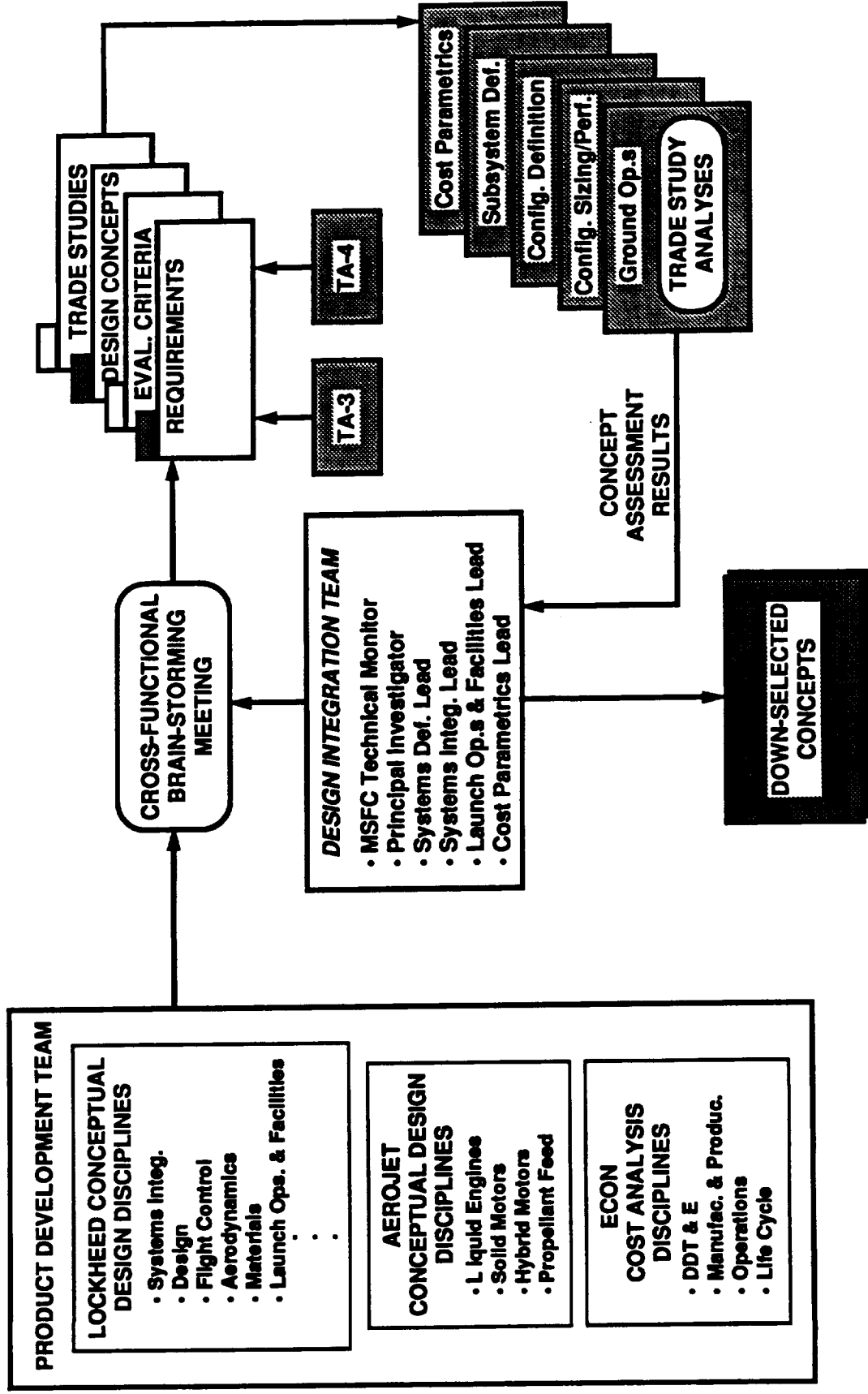
TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92

ADVANCED TRANSPORTATION SYSTEM STUDIES TA-2 HEAVY LIFT LAUNCH VEHICLE STUDY



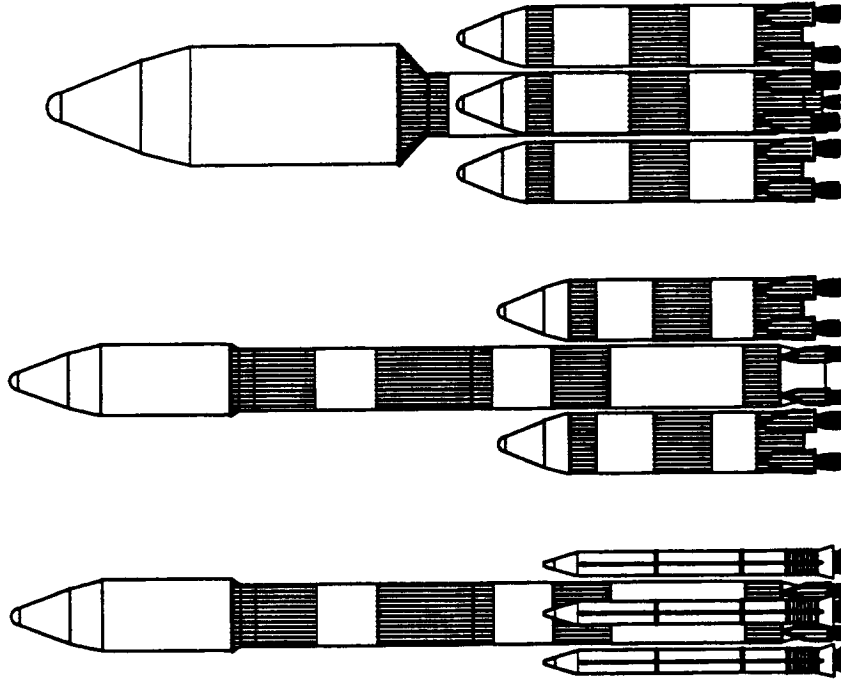


STUDY APPROACH





TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92



First Year Schedule



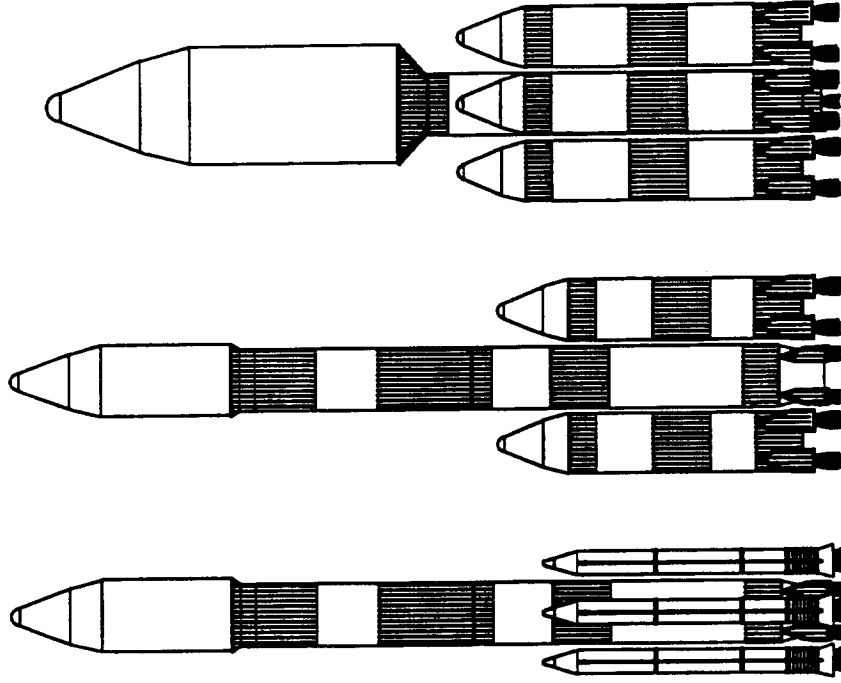
J. B. McCurry
713-333-6579



First Year Task Activity Schedule

ADVANCED TRANSPORTATION SYSTEM STUDIES TA-2 FIRST YEAR (BASIC)

Activities		1	2	3	4	5	6	7	8	9	10	11	12
1. Sys. Requirements and Selection Criteria...		 (1)											
2. Vehicle Design		 (2)											
Configuration Conceptual Definition		 (6)											
Propulsion Subsystem Conceptual Definition		 (3)											
Vehicle Sizing & Performance Evaluation		 (4)											
Parametric Engineering Cost Analysis		 (5)											
3. Ground Operations & Facility Evaluations													
4. Technology Requirements Definition		 (7)											
5. Concept Selection and Evaluation		 (8)											
6. Concept Baseline/Evolution		 (9)											
ATSS Integration Group													
Proposals for Option Studies													
Customer Reviews		   											
Status Reports		   											
Formal Reports		   											
NOTE: (n) = PRIORITY WITHIN THE YEAR													



Major Tasks



Major Tasks

First Year

- Task 1: Systems Requirements and Selection Criteria Definition
- Task 2: Vehicle Design (Config. Definition, Propulsion Definition, Sizing and Performance Evaluation, Parametric Cost Analysis)
- Task 3: Ground Operations and Facility Evaluations
- Task 4: Technology Requirements Definition
- Task 5: Concept Selection and Evaluation
- Task 6: Concept/Baseline Evolution

Second Year

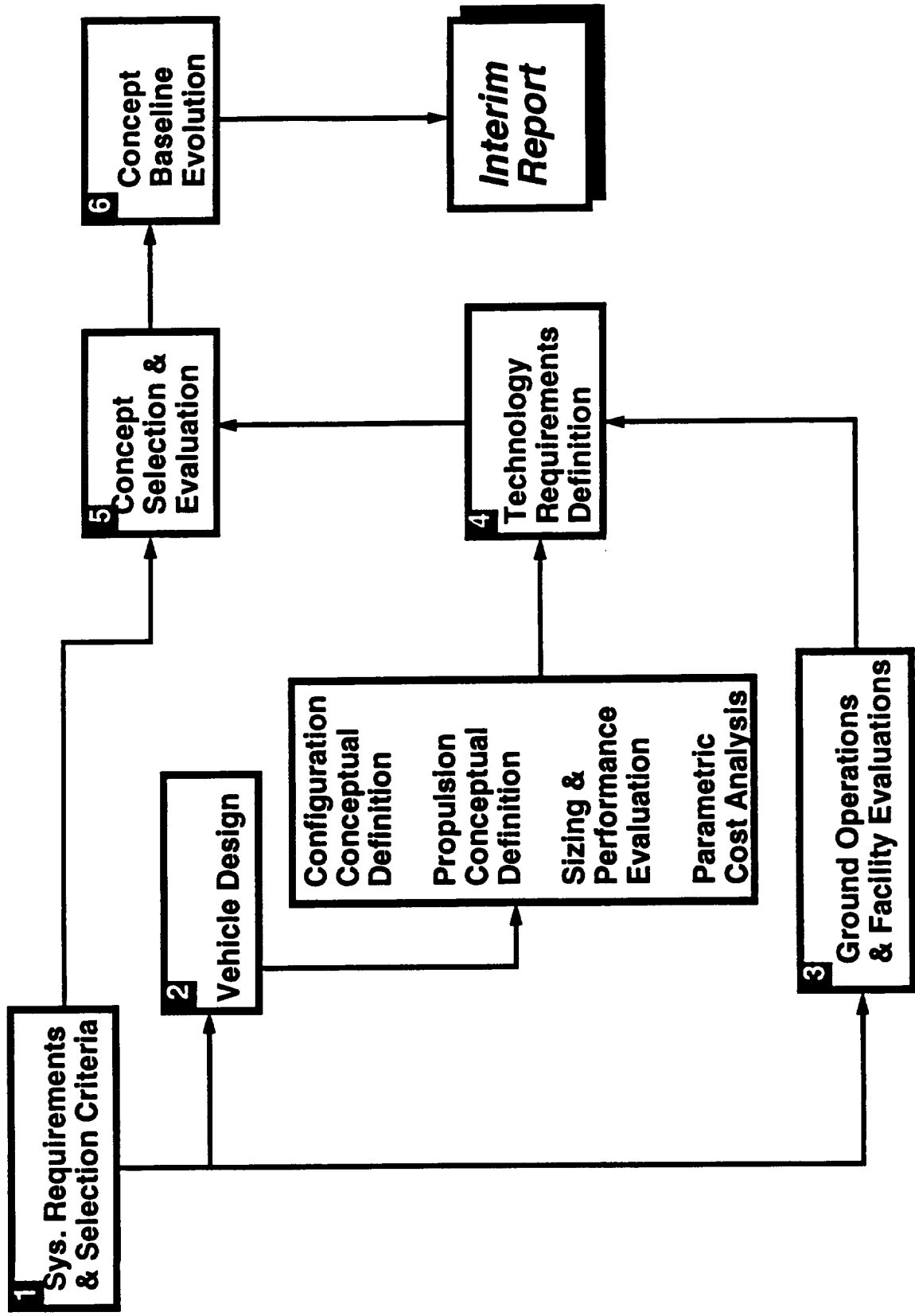
- Task 1: Requirements and Mission Scenario Update
- Task 2: Vehicle Design
- Task 3: Launch Site Evaluations
- Task 4: Technology Requirements Assessments

Third Year

- Task 1: Vehicle Design
- Task 2: Manufacturing Facility Definition
- Task 3: Launch Site Assessments
- Task 4: Technology Development Study
- Task 5: Environmental Impact Assessment Report



First Year Task Flow

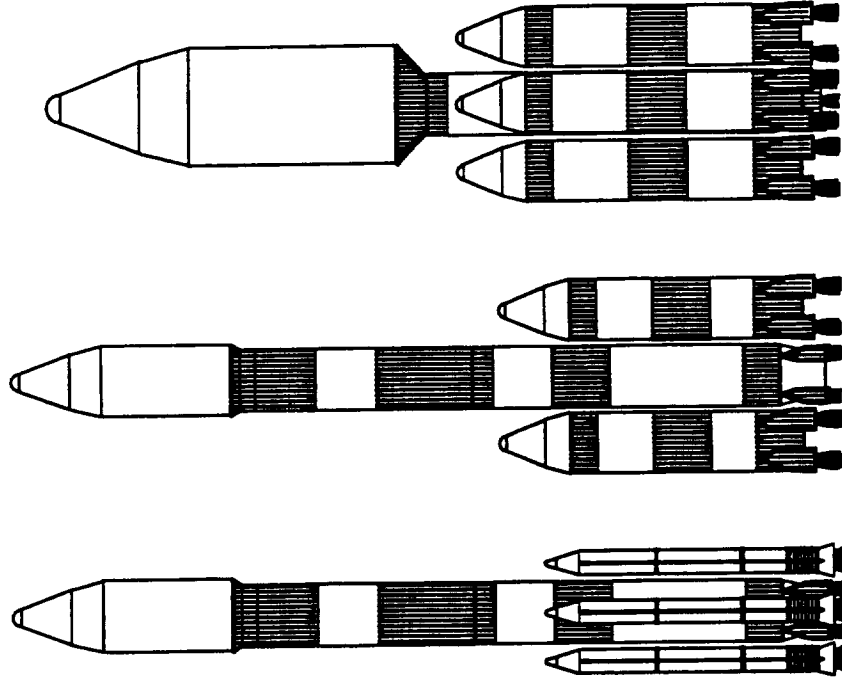




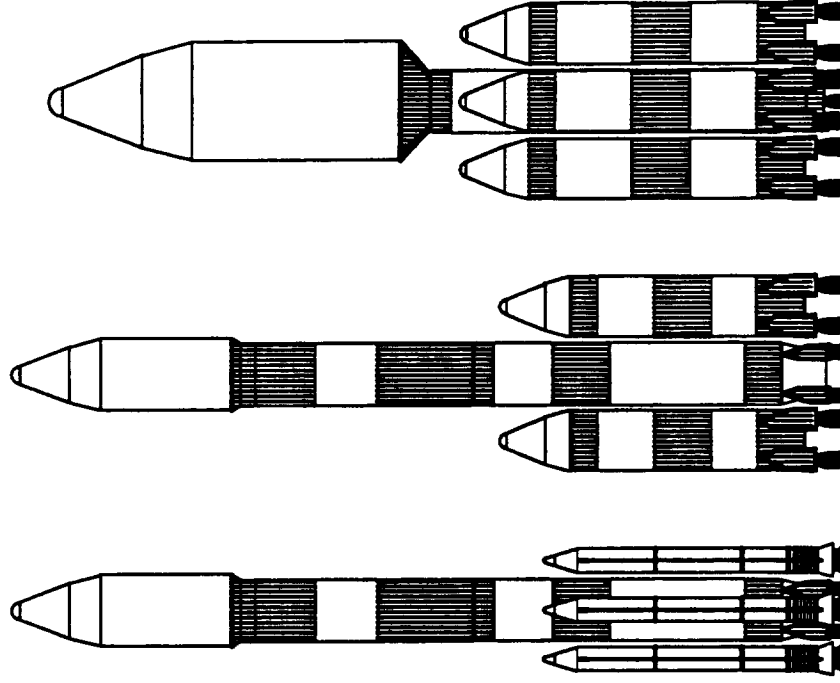
First Year Task Resource Summary

<u>TASK</u>	<u>MAN-HOURS *</u>
1 System Requirements and Selection Criteria	670
2 Vehicle Design	2100
3 Ground Operations and Facility Evaluations	2320
4 Technology Requirements Definition	380
5 Concept Selection and Evaluation	390
6 Concept/Baseline Evolution	670
Interim Report	760
Total	7290

* 151 MAN-HOURS = 1 MAN-MONTH



3. Study Methodology



Vehicle Design and Performance Assessments

M. Berning/Lockheed



- **Trajectory Analysis and Performance Optimization Codes**
 - Simulation and Optimization of Rocket Trajectories (SORT)
 - Optimal Multi-Arc Trajectory Program (OMAT)
- **Vehicle Sizing and Mass Estimation Codes**
 - Mass Fraction and Empirical Weight Estimation Codes built into the SORT program
 - Mass Fraction and Empirical Weight Estimation Codes developed in PC applications such as TK! Solver and Lotus 1-2-3
 - Launch Vehicle Geometry Sizing Codes developed in TK! Solver



SORT BACKGROUND

- Simulation and Optimization of Rocket Trajectories (SORT)
- Generalized multipurpose 3 Degrees-of-Freedom trajectory simulation with 3 axis moment balance
- Generalized multi-level parameter optimization
- Developed in 1984 to streamline Space Shuttle Ascent Flight Design
- Public Domain Software (COSMIC)

SORT USERS

- Aerojet (Iuka, Mississippi)
- Barrios Technologies Inc.
- Eagle Engineering
- EER Systems
- Lockheed Engineering and Sciences Company (Houston)
- Lockheed Missiles and Space Company
- McDonnell Douglas Space Systems Company
- NASA JSC
- Rockwell Space Operations Company (RSOC)
- Space Industries



COMPUTERS/OPERATING SYSTEMS

- Cray/UNICOS
- HARRIS 1200
- HP9000/UNIX
- IBM Compatible PCs/DOS
- Macintosh II/Apple OS
- Pyramid/UNIX
- Sun Workstations
- UNISYS 1100 Series
- VAX Workstations and Mainframes

DOCUMENTATION

- User's Guide
- Programmer's Guide
- Installation Procedures
- Programmer's Aids/Development Tools Manual



6 DOF Spinoffs of SORT

- Adaptive Guidance Throttle (AGT) I-load Design Processor
- Day-of-Launch I-load Biasing System (DIBS)
- Engine-Out Guidance and Control I-load Processor

LAUNCH VEHICLES MODELED WITH SORT

- Air-Launched PLS
- Arianne
- CERV
- Flyback Booster
- HLLV
 - Clean Sheet Concepts
 - NLS Derived Concepts
 - Saturn Derived Concepts
- NLS
- PLS
- Saturn V
- Shuttle C
- Shuttle II



LAUNCH VEHICLES MODELED WITH SORT(continued)

- Space Shuttle
- Titan 34D
- Titan III
- Titan IV

TYPICAL APPLICATIONS

- Ascent Performance Assessments for Numerous Launch Vehicles
- Crew Escape Module Sizing Analysis
- Entry Trajectory Analysis
- Advanced Guidance Algorithm Test Bed
- Launch Vehicle Sizing
- Nominal Guidance and Control I-load Design for the Space Shuttle
- Orbit Insertion I-load Design for the Space Shuttle
- Solid Rocket Motor Thrust vs Burn Time Optimization



PROGRAM CAPABILITIES

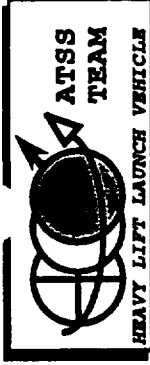
- Aerodynamic Modelling
 - Lateral and Longitudinal Forebody Forces and Moment Coefficients
 - Longitudinal Base Forces and Moments Coefficients
 - Ballistic Coefficient Drag Model
- Atmosphere Models
 - Generalized Table Lookup Model
 - 1962 U.S. Standard Atmosphere
 - 1963 Patrick Air Force Base Atmosphere
 - Generalized Exponential Atmosphere Model
- Discrete Event Simulation Model
 - Fixed or floating events
 - Unlimited choice of interrupt variables



- Engine/Propulsion Model
 - Generalized Table Lookup Thrust, ISP, Flowrate, Exit Area Tables
 - Fixed/Controllable Engines
 - Throttleable/Non-throttleable engines
 - Single/Multiple Throttle Controllers
 - Monopropellant or bipropellant engines
 - Precise 3 axis static moment balance
 - Precise sensed acceleration limiting
 - Solid Rocket Motor Propellant Mean Bulk Temperature (PMBT) Adjustment Model



- Guidance/Steering Algorithms
 - Various Euler Reference Frames using any user specified rotation sequence
 - LVLH
 - Plumbline
 - Boost Reference
 - Aerodynamic Attitude Steering
 - Mixed Attitude Angle Steering
 - Open Loop Steering Algorithms
 - 3rd order polynomial steering coefficients
 - Piecewise continuous steering commands
 - Generalized table lookup attitude tables
 - Rate limited attitude commands
 - Closed Loop Guidance/Steering Algorithms
 - Aerodynamic load steering
 - Linear Error Rate Feedback
 - Linear Tangent Exoatmospheric Guidance (PEG)
 - Numerical Iterative Steering



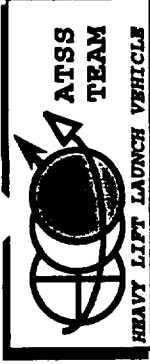
- Mass Tracking Model
 - Individual Mass Subsystems
 - Individual Subsystem Center of Gravity Computation
 - Vent/Ablation Model
- Numerical Integration Algorithms
 - Standard 4th Order Runge Kutta
 - Adams Moulton Predictor/Corrector
 - Variable Step Size Runge Kutta Fehlberg (4th order with 5th order step size control)



- Multi-level Parameter Optimization
 - Seven levels of nested iteration/optimization
 - Up to 20 control variables per iteration level
 - Up to 20 dependent variables per iteration level
 - Unlimited choice of control variables including event termination criterion
 - Unlimited choice of target/constraint variables
 - Unlimited choice of optimization criterion
 - Various iterators/optimization codes
 - Newton Raphson
 - Secant Method
 - Quadratic Search
 - Parametric Scan
 - J.B. Rosen's Projected Gradient Algorithm (nearly identical to the optimization algorithm in the POST program)
 - Variable Metric Algorithm for Constrained Optimization (VMACO) (based upon J.D. Powell's VFO2AD algorithm)
 - NPSOL Algorithm (Systems Optimization Laboratory, Stanford University; available in Version 7.0, October '92)



- Sizing/Weight Estimation Algorithms
 - Series or Parallel Burn Vehicles
 - Generalized Stage Mass Fraction Option
 - Empirical Weight Buildup by Vehicle Subsystems
- Miscellaneous Models
 - Aerodynamic Stagnation Heating Model
 - Chapman's Heating Model
 - Detra-Kemp-Riddell
 - Shuttle/ET Heating Indicator
 - Booster Forward Attachment Shear Loads Model
 - Generalized Integration Variable Model
 - Parameter maximum/minimum monitoring
 - Vacuum impact point predictor
 - Velocity Loss Model
 - Vertical Hold Down Model
 - Wing Shear, Bending and Torsion Model



- Input/Output
 - Standard ASCII Text Files for Input and Output
 - Mixed Units (SI or English) for Input
 - SI or English Units for Output
 - Cumulative Iteration Summaries
 - Single or Multiple Input Files



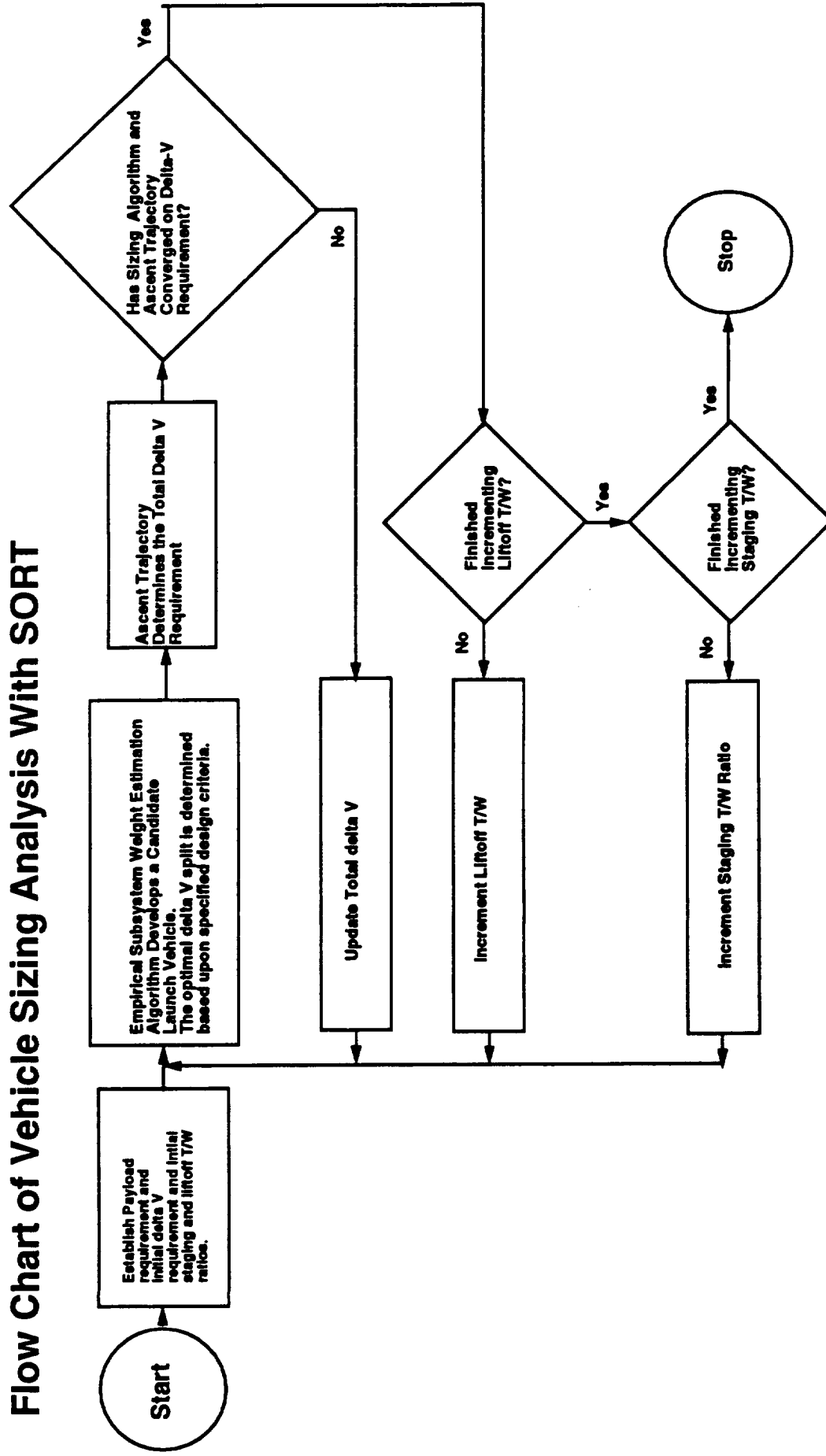
LAUNCH VEHICLE SIZING CAPABILITY

- NLS/SATURN DERIVED METHODOLOGY
 - Stage Mass Fraction Based Sizing Algorithm
mf = f (thrust, propellant load, etc.)
- CLEAN SHEET APPROACH
 - Mass Subsystem component buildup based primarily upon the work of MacConchie and Klich as well as historical data compiled by JSC
 - Engine System
 - Thrust Structure
 - Pressurization and Feed Systems
 - Actuation Subsystems
 - Electrical Subsystems
 - Hydraulics/EMAs
 - Primary Power Systems
 - Interstage Weights
 - Propellant Tankage
 - Unusable Propellants
 - Flight Performance Reserves
 - Growth Margin



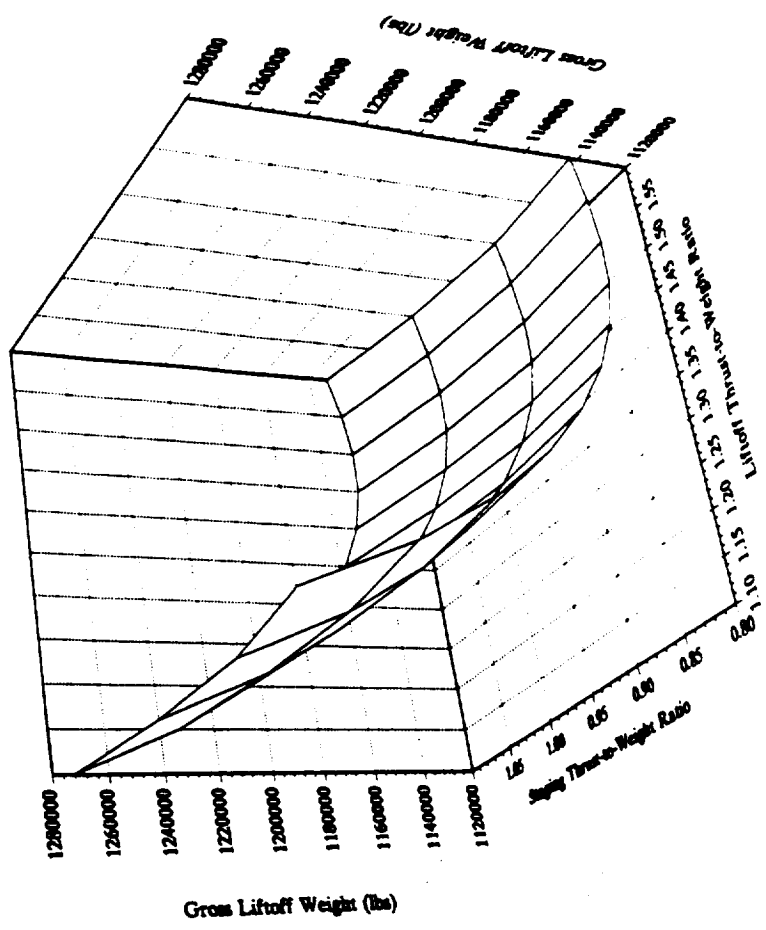
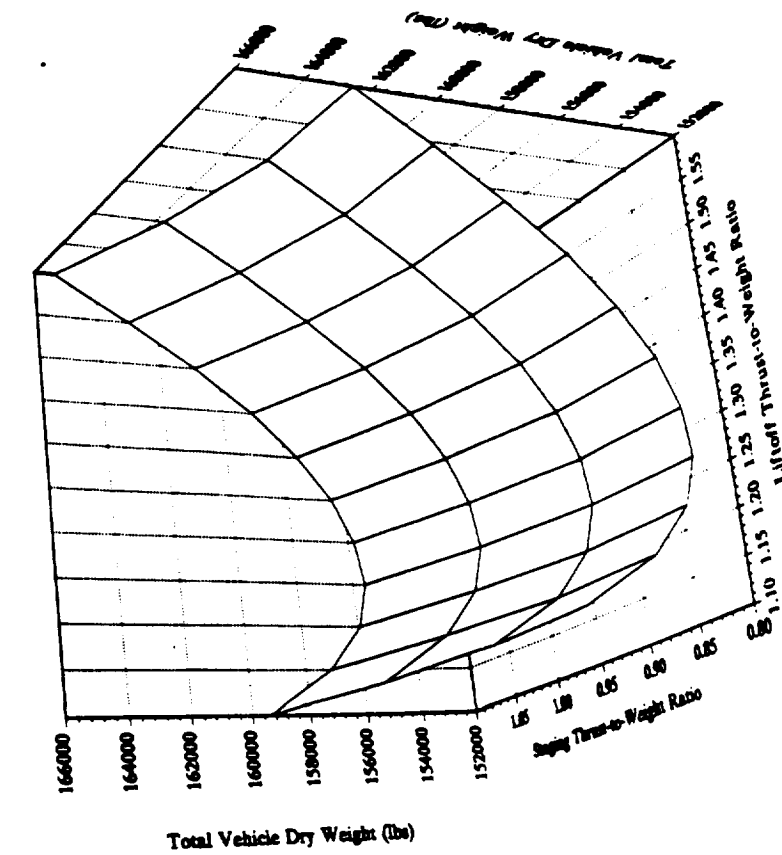
- Engine Thrust Options
 - Fixed initial T/W; fixed number of engines; variable thrust
 - Fixed initial T/W; variable number of engines; fixed thrust per engine
 - Variable initial T/W; fixed number of engines; fixed thrust per engine
- Staging Delta V Split Optimization Options
 - Fixed Delta V for Booster Stage
 - Minimum Gross Liftoff Weight
 - Minimum Dry Weight
 - Propellant Tank Commonality
- Engine Out Protection Design Options
 - Extra Engine per stage
 - Extra Propellant
 - No Engine Out Protection

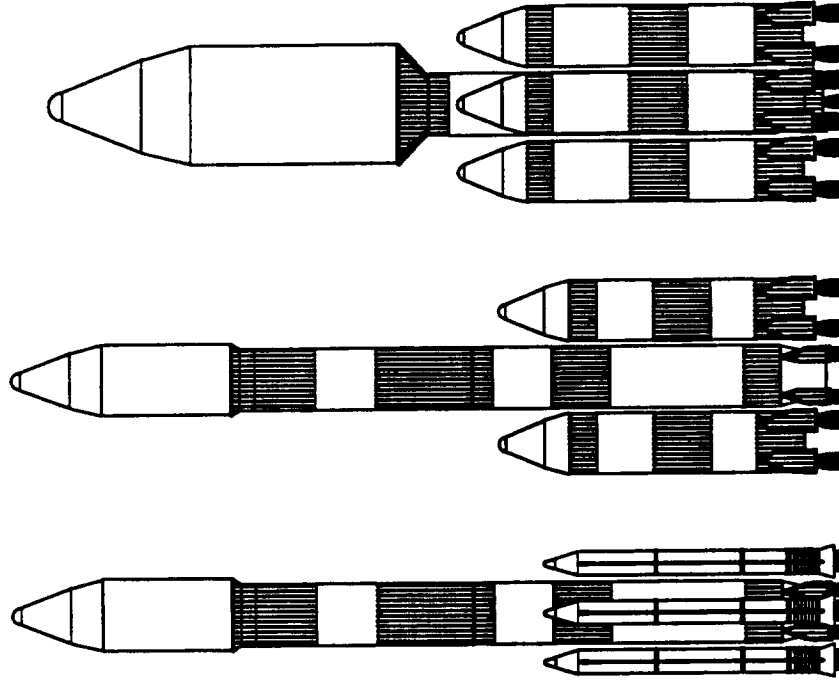
Flow Chart of Vehicle Sizing Analysis With SORT





Representative Two Stage 50 Klbf Payload Launch Vehicle Sensitivity to Liftoff and Staging T/W Ratios





Analysis Tool Benchmarks

K. Sagis/Lockheed



Introduction

- Regression testing with previously verified tools is most common method of simulation verification.
 - .. SORT versus POST comparison
 - .. SORT versus MASTER and ASTRO comparison



POST Testcase Description

Mass Properties

First Stage Inert (lbf)
First Stage Propellant (lbf)
Second Stage Inert (lbf)
Second Stage Propellant (lbf)

308,000
809,000
665,000
2,249,000

Miscellaneous

ETR Launch
1962 Standard Atmos.
No Winds
50 x 106.5 nm Orbit

Propulsion

First Stage lsp (sec)
First Stage Thrust (lbf)
First Stage Exit Area (ft²)
Second Stage lsp (sec)
Second Stage Thrust (lbf)
Second Stage Exit Area (ft²)

439.0
5,472,000
232.5
459.0
1,431,000
154.54



Trajectory Summary

Lift-off

GLOW (lbf) 4,033,274 (4,033,200)
T/W 1.235 (1.235)

Maximum Dynamic Pressure

Time (sec) 75.0 (75.0)
Dynamic
press. (psf) 422.5 (430.9)

Staging

Time (sec) 180.45 (180.45)
Altitude (nm) 28.78 (28.26)
Weight (lbf) 1,784,274 (1,784,200)
Velocity (fps) 7,646.8 (7,657.2)
Flight path
angle (deg) 13.9 (13.7)

MECO

Time (sec) 459.26 (459.26)
Altitude (nm) 50.00 (50.00)
Weight (lbf) 310,274 (310,200)
Velocity (fps) 25,853.1 (25,853.1)
Flight path
angle (deg) 0.000 (0.000)

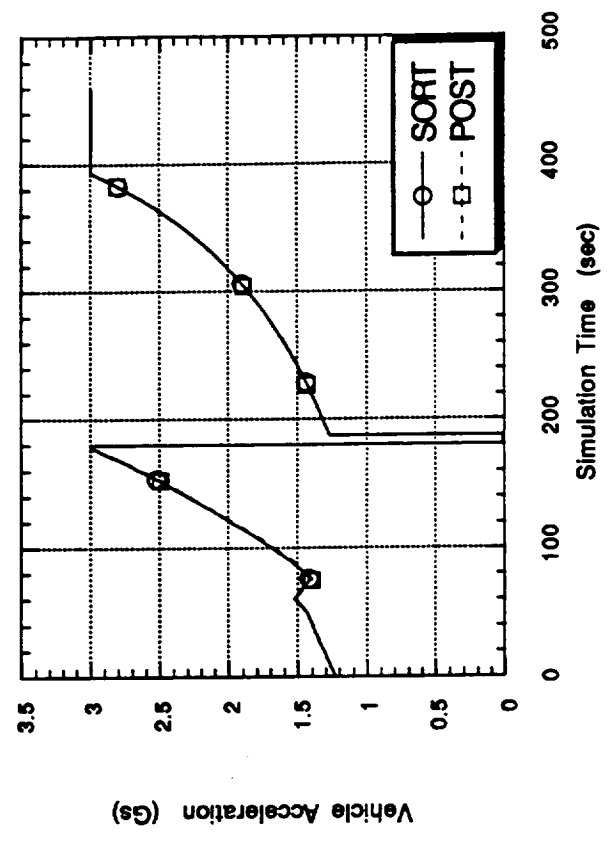
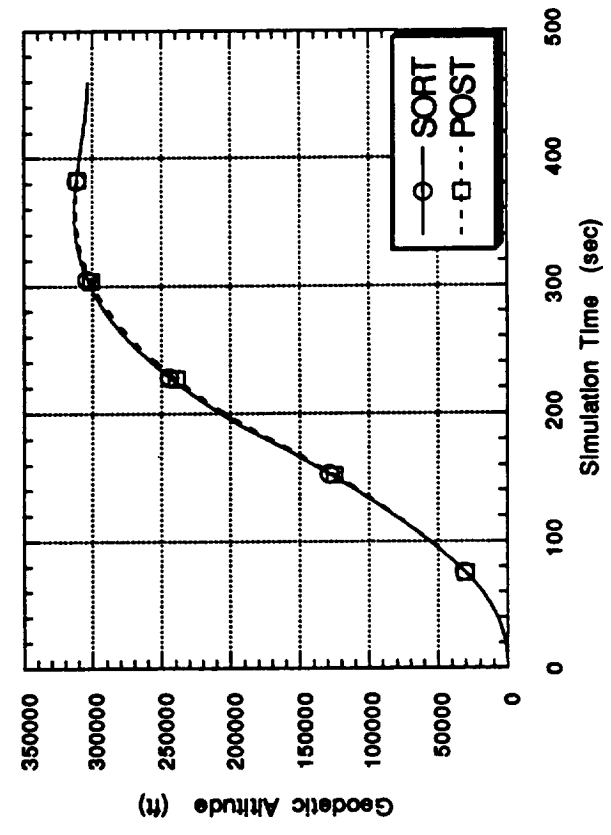
Legend

SORT Results (POST Results)



Plot Comparisons

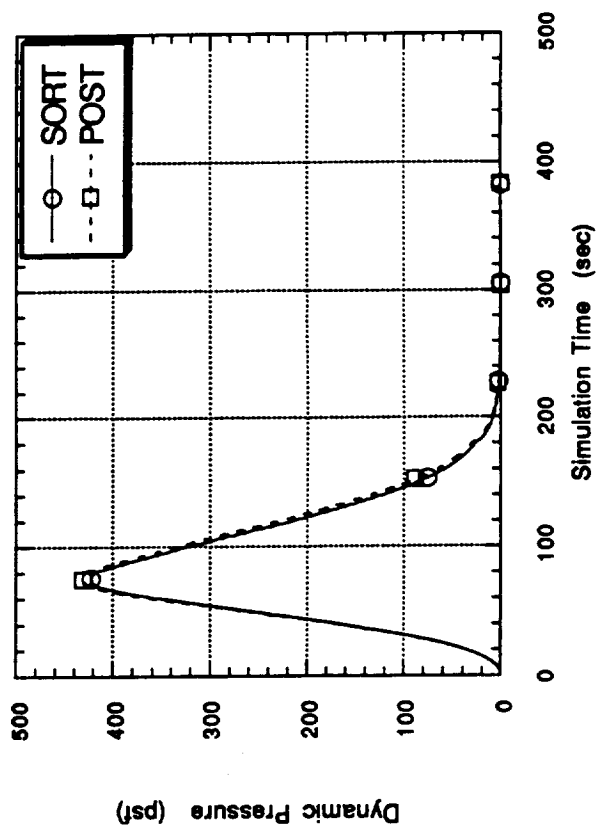
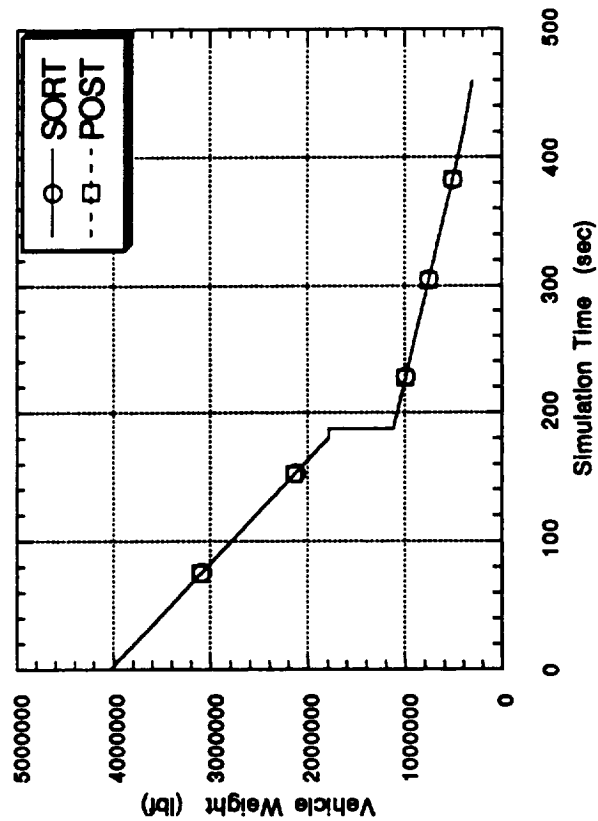
Altitude and Acceleration

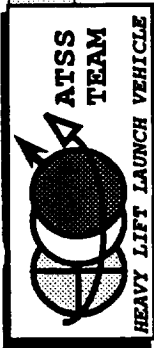




Plot Comparisons

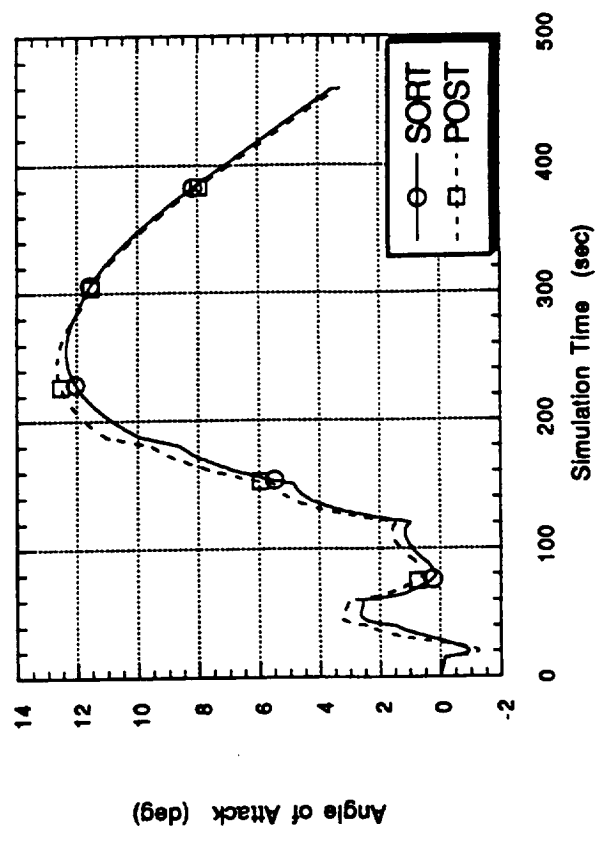
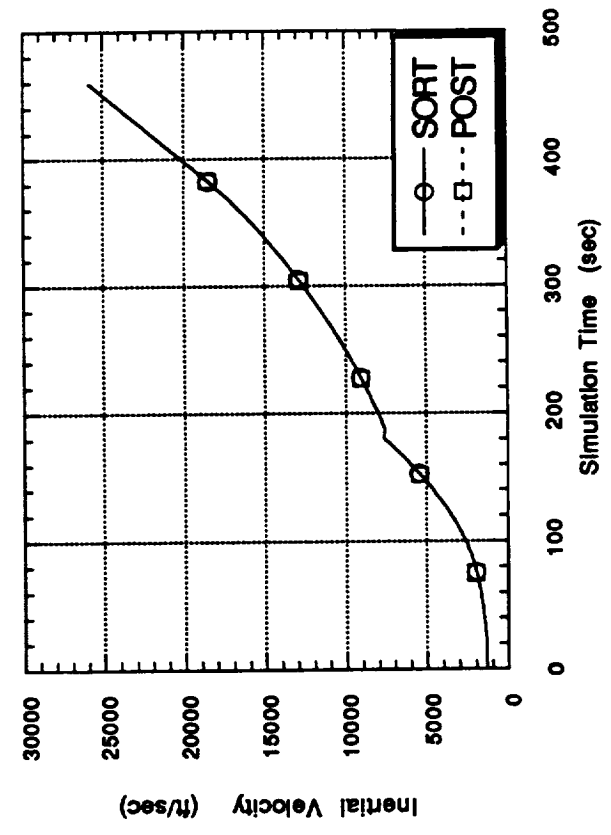
Weight and Dynamic Pressure

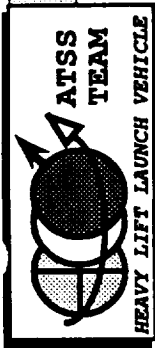




Plot Comparisons

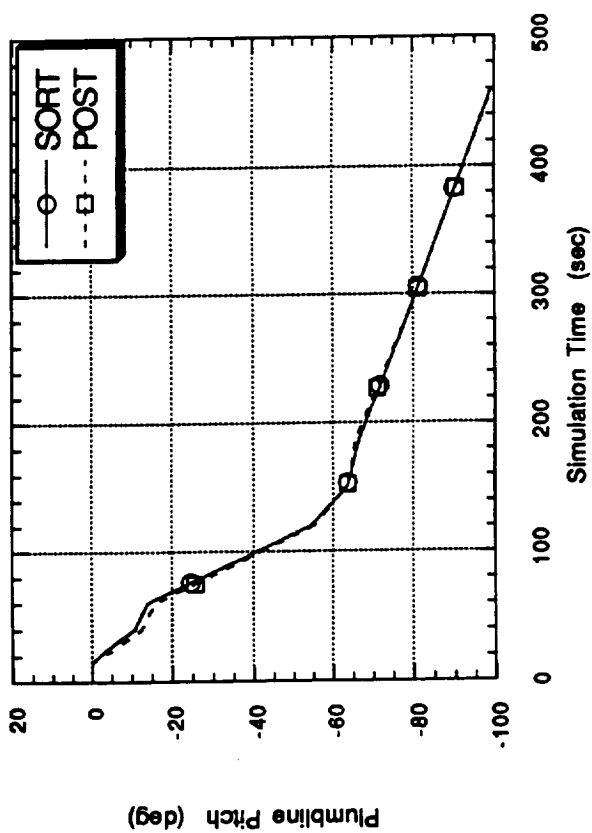
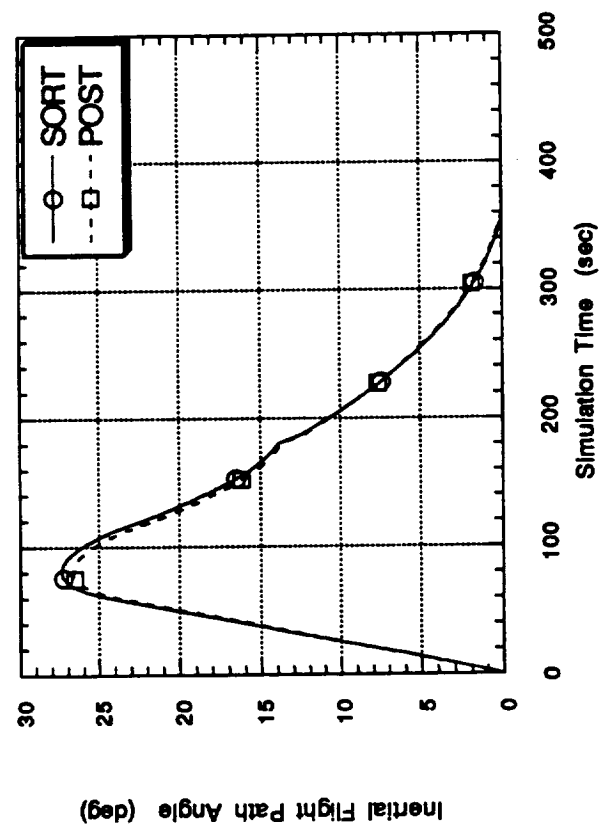
Velocity and Angle of Attack





Plot Comparisons

Flight Path Angle and Plumbline Pitch





National Launch System Cycle 0 Simulation Comparison

Lockheed	MSFC EP55	Rockwell International
----------	--------------	---------------------------

NLS 1

Engine Out	5,888	6,343	5,999
Nominal	14,214	14,687	14,133

NLS 2

Engine Out	-2,283	-2,741	-3,404
Nominal	7,518	7,878	7,476

Note: Values given denote excess
MPS propellant (lbf) at MECO

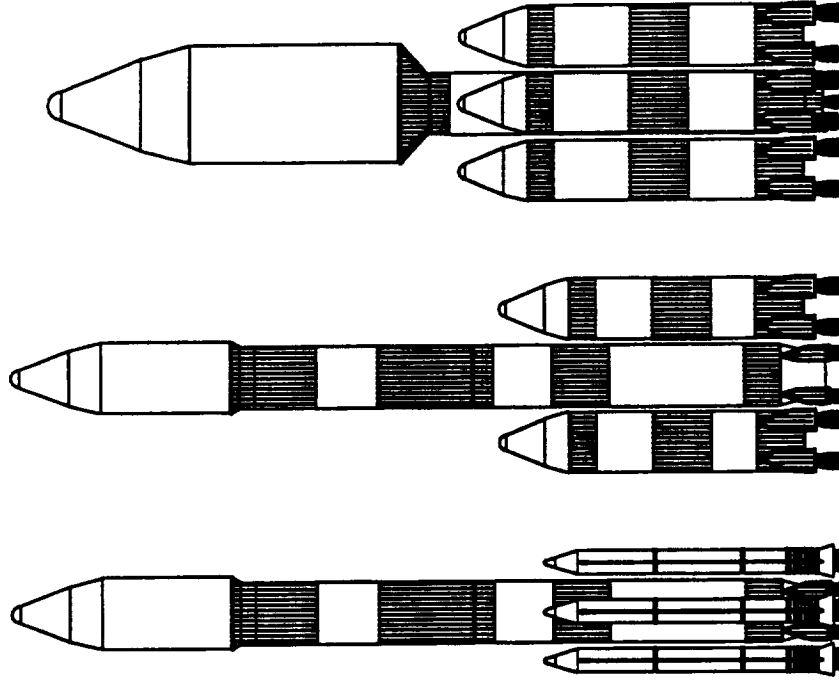


Summary and Recommendations

- SORT agrees with the industry standards for trajectory analysis.
- A trajectory design data package (TDDP) and simulation comparison should be developed for SEI class vehicles to calibrate ATSS participants.



TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92



Ground Operations Assessments

G. Letchworth/Lockheed



G. F. Letchworth
407-383-2200, X-2829



TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92

Study Methodology— Ground Operations

Generate Top-Level Operations Concepts and Scenarios

- Depicts Major Movements of HLLV Flight Elements/Integrated Vehicle through Launch Facilities
- Overall Approach for Processing Launch Vehicle (ITL vs IOP)
- Flight Element Arrival to Launch



Study Methodology— Ground Operations

Analyze Launch Vehicle Processing Requirements

- Flight Element Processing Requirements and Resulting Timelines
- Integrated Vehicle Processing Requirements and Resulting Timelines
- Single Flow— Select HLLV Configurations



Study Methodology— Ground Operations

Analyze Payload Integration Requirements

- Top-level Payload Integration Concepts/Scenarios
- Payload Integration Requirements and Resulting Timelines
 - Encapsulation of Cargo Elements
 - Launch Vehicle Integration
 - Pad Servicing
- Select HLLV Configurations
- "Offline" Payload Processing Not Addressed



Study Methodology— Ground Operations

Assess Mixed Fleet Operations

- Multi-flow Processing Schedules
- Dependant on Launch Site
- Dependant on Number and Type of Vehicles at Site
- Dependant on Launch Vehicle Flight Rates/Launch Intervals
- One or Two Launch Sites/HLLV Configurations
- Evaluates Facility Bottlenecks
- Evaluates Launch Site Total Throughput Capability



Study Methodology— Ground Operations

Generate Facility Concepts and Requirements

- Identify Top-level Facility Requirements to Support Processing
 - Flight Elements
 - Integration
- Facility Sizing Dependant on Individual and Mixed Fleet Schedules
- Conceptual Facility Layouts
- New vs. Existing Facility Utilization Dependant on Launch Site
- Construction and Activation Timelines
- Select HLLV Configurations



Study Methodology— Ground Operations

Produce Operations Concept Document

- Narrative of HLLV Launch Site Processing Flow
- Flight Element Descriptions
- Facilities Descriptions
- Narrative of Processing Steps
 - Standalone Flight Elements
 - Integrated Vehicle
- One or Two HLLV Configurations



TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92

Study Methodology— Ground Operations

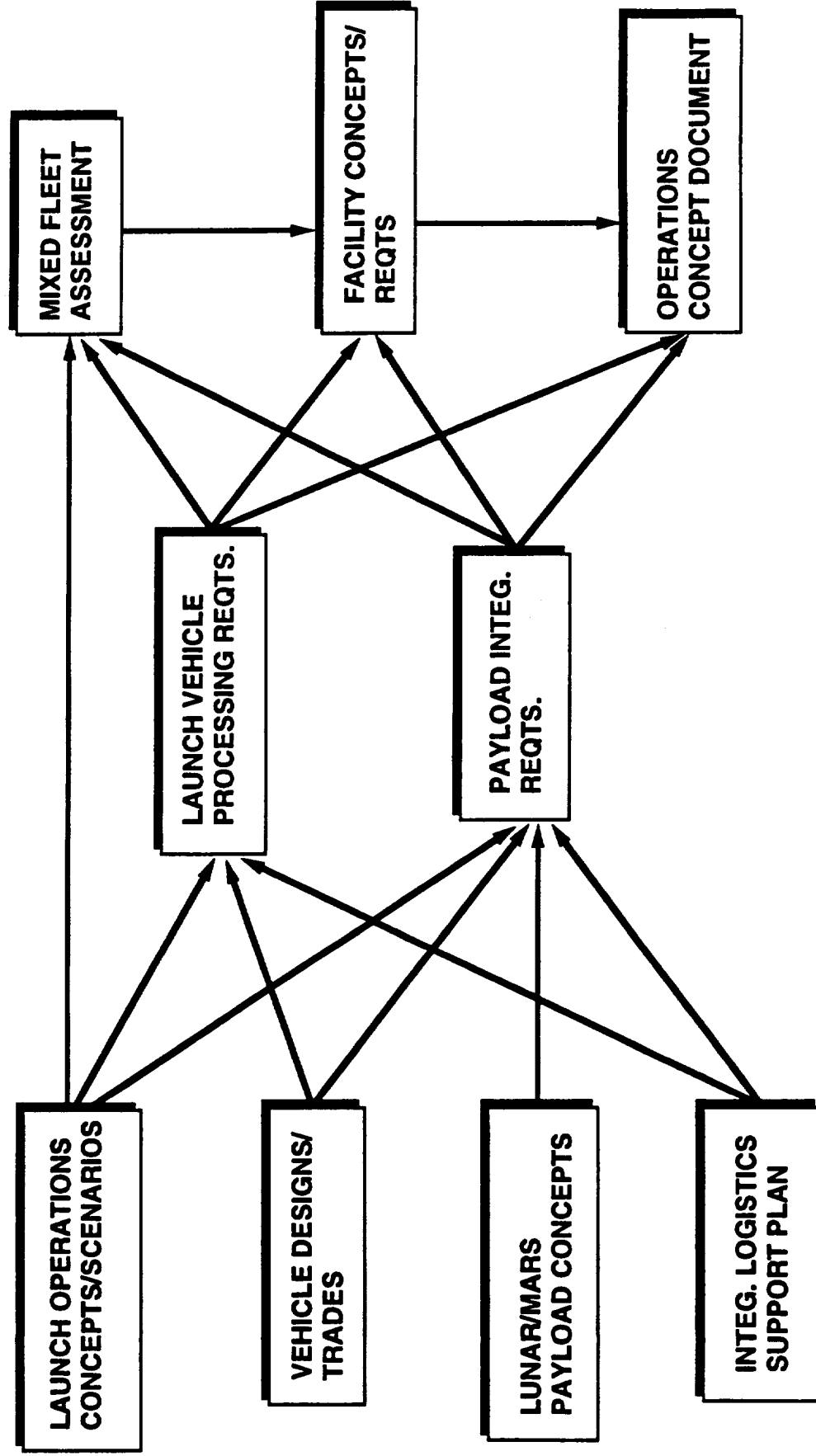
Generate Integrated Logistics Support Plan

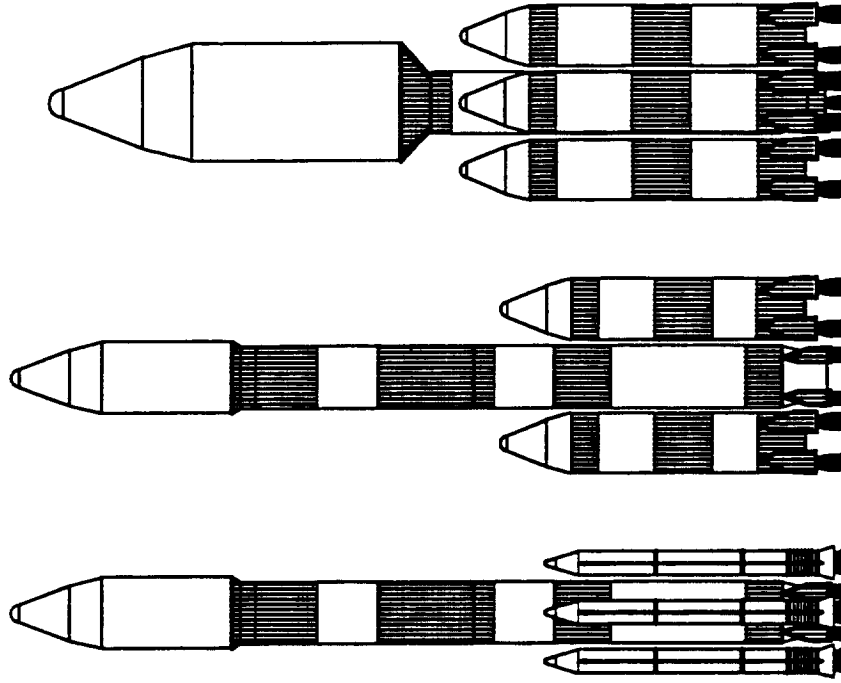
- Top-level Draft Document
- Logistics Support Analysis
- Maintenance Concept
- Acquisition Strategy
- Standardization Plan
- Reliability, Availability and Maintainability





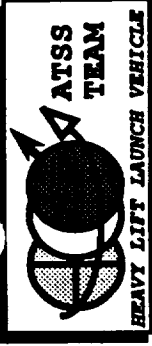
Study Methodology— Ground Operations





Parametric Cost Analysis

J. Skratt/ECON



ECON EXPERIENCE IN SPACE TRANSPORTATION COST ANALYSIS

- ECON has 19 years experience in the cost and economic analysis of current and future space transportation systems
- Recent studies and analyses include :
 - Space Transportation System
 - STAS, ALS and NLS
 - Shuttle - C
 - ACRV
 - PLS - HL 20 configuration
 - NASP Derived Vehicles
 - Hybrid Propulsion Systems
 - SSTO
- Clients have included NASA HQ, MSFC and JSC, the USAF Space Division and Commercial Companies
- ECON has worked with or for companies associated with each of the Technical Areas of the ATSS

ECON, INC.
JOHN SKRATT
(408) 249-6364



REQUIREMENTS FOR PROVIDING INTELLIGENT COST ESTIMATES

- Skilled Cost Analysts with experience in launch vehicle design
- General Case Tools that support a level of detail consistent with design assumptions
- A database of cost information on relevant historical launch systems, subsystems and components
 - Database must be organized in such a way that the cost models can interpret the principal assumptions associated with each data point
- An explicit recognition of the uncertainties in the design and cost estimation process
- An ability to reconcile and/or defend estimates in light of other cost analyses
- An integrated cost/design team
 - Management objective
 - Electronic integration
 - Rapid analyses with output that supports subsequent design iterations

ECON, INC.
JOHN SKRATT
(408) 249-6364



FEATURES OF ECON'S APPROACH TO COSTING FUTURE SPACE LAUNCH SYSTEMS

- Technology Forecasting
 - Tech-Up and Tech-Down
 - Analytical method for explicitly linking the time dependent technology levels of the historical analog and projected new design
 - Calibration variable similar to the technique employed by PRICE
 - Works at the component, subsystem and system levels
- New Ways of Doing Business
 - Includes Development, Manufacturing, Procurement Practices, Organizational Variations and Cultural Changes
 - First, "design" the changes and identify the pertinent assumptions
 - Then determine the range of cost savings to a specific system that could be derived from such changes
 - Utilize models and databases with suitable levels of detail to estimate the savings within the boundaries already established
 - This is an assessment in which a thorough identification of the assumptions is more important than the estimated values

ECON, INC.
JOHN SKRATT
(408) 249-6364



FEATURES OF ECON'S APPROACH TO COSTING FUTURE SPACE LAUNCH SYSTEMS - Cont'd

- Launch Vehicle Operations
 - ECON has developed and implemented a simple top-down parametric model of expendable launch vehicle operations based on historical systems ranging from the Scout to the Titan IV
 - Input variables include the GLOW, number of engines in the stack and the average unit production cost
 - The model is sensitive to annual flight rate and is "disaggregated" down to six elements of operations
- Uncertainty Estimates
 - Any estimate of the cost of a future system has uncertainty associated with it
 - Presentation of these uncertainties allows the decision maker to quantify the acceptable level of risk by picking a value from the distribution
 - While the basis of uncertainty estimates is often subjective it is better than a deterministic answer and can be used to discriminate between future systems that are "risky" for different reasons

ECON, INC.
JOHN SKRATT
(408) 249-6364

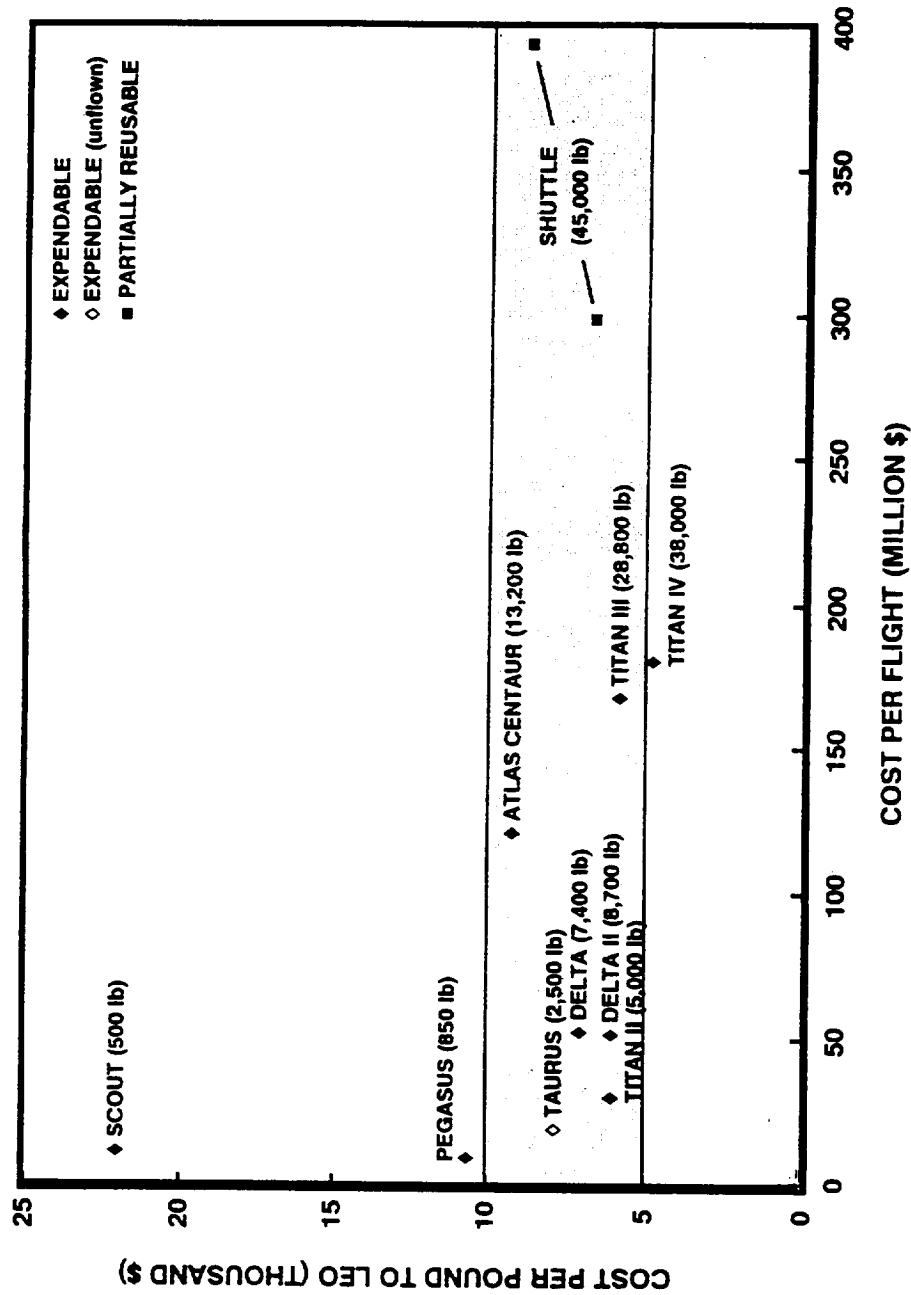


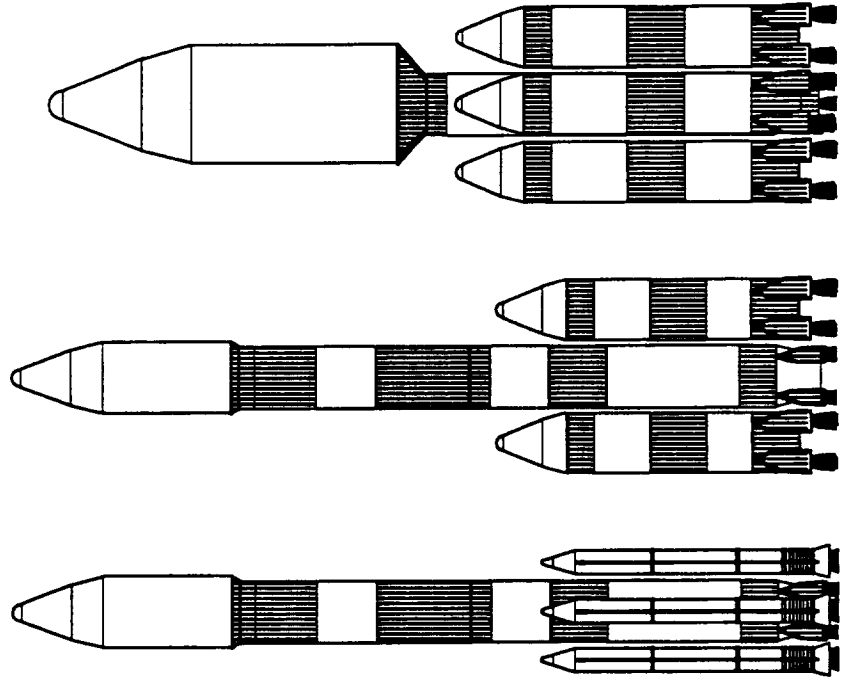
PROPOSED COST ESTIMATING TASKS FOR THE FIRST YEAR OF TA-2

- **Coordinate costing groundrules for TA-2, including the WBS, formats for results and cost objective functions**
- **Assemble / develop Launch Vehicle cost database appropriate to the Saturn V derived and Clean Sheet HLLV options at a subsystem level**
- **Modify existing Acquisition Cost Models as necessary based on the designs and the output requirements of the study**
- **Model Acquisition and operations costs for The HLLV options using top-down parametric approaches**
- **Support HLLV requirements and subsystem trade studies**
- **Participate in an ATSS Cost Analysis Working Group established for the purpose of coordinating all cost groundrules and the peer review and coordination of all cost estimates**



TRENDS IN LAUNCH VEHICLE COST EFFICIENCY





Subsystem Definition and Technologies

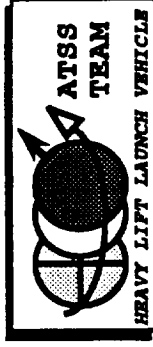
L. Rustenburg/Lockheed





AGENDA

- Disciplines/Facilities
- Capabilities of particular interest to MSFC
- Conclusion



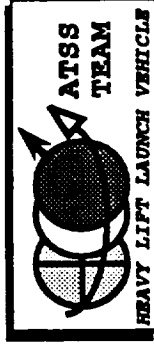
DISCIPLINES

- **ENGINEERING TECHNOLOGY & TEST ENGINEERING**
 - **Aerothermodynamics, fluid & structural mechanics, dynamics analysis and simulation**
 - **Marine Systems**
 - **Mechanical Engineering**
 - **Avionics Engineering**
 - **Propulsion Systems Engineering**
 - **Electronic Systems Engineering**
 - **Fleet Ballistic Missile Systems Engineering & Integration**
 - **Advanced Projects Engineering**



DISCIPLINES

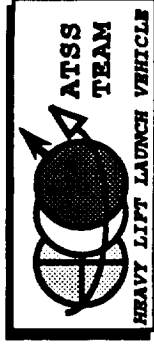
- ENGINEERING TECHNOLOGY & TEST ENGINEERING (CONTINUED)
 - Engineering & Test Labs
 - Acoustic Test Facility
 - Thermal-Vacuum Facility
 - Structural Test Facility
 - Environmental Test Lab
 - Modal Test Facility
 - Material & Processes Lab
 - Manufacturing Facilities
 - Electronics Manufacturing
 - Santa Cruz Test Base Facility



SELECTED CAPABILITIES

MISSILE SYSTEMS DIVISION

- **ENGINEERING**
 - Vehicle design
- **PRODUCTION & OPERATIONS**
 - Advanced Development Engineering



VEHICLE DESIGN

CAPABILITIES

- **Metallic & composite structure design/analysis**
- **Separation system design/analysis**
- **Concept development & production engineering**
- **Subsystem/component testing**
- **Smart structure development**
- **Electronic/hard mock-up**
- **Subsystem integration**
- **Product Development Team (PDT) environment**
- **Rapid prototyping (stereolithography)**



VEHICLE DESIGN

ELECTRONIC MOCK-UP

- Packaging/subsystem integration
- Vehicle/facility interface
- Assembly sequence evaluation
- Vehicle transportation/handling study



VEHICLE DESIGN

RESOURCES

- **COMPUTERS**
 - 2D/3D solid modelling CAD/CAE systems (CADAM, CATIA, PRO-ENGINEER, INTERGRAPH, IDEAS)
- **LABS**
 - Structural test lab
 - Mechanical control lab with tie-in to mechanical simulation lab
 - Santa Cruz test base for propellant and ordnance testing
 - Hard and soft mock-up
 - Engineering lab for material & process development & hardware demo
- **PERSONNEL EXPERIENCE**
 - NASA man rated design requirements
 - PDTs for advanced structures, mechanical controls, advanced material applications
 - Detail hardware design, analysis, fab, assembly & procurement of major systems



VEHICLE DESIGN

PROGRAM EXPERIENCE

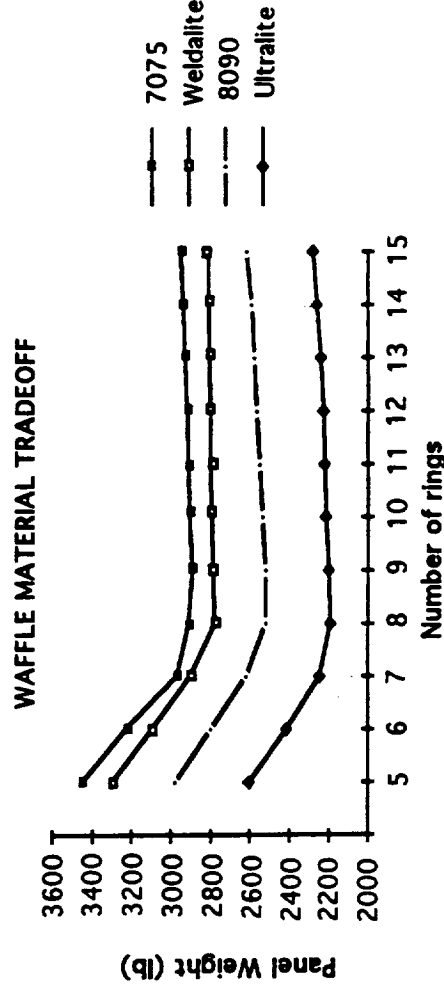
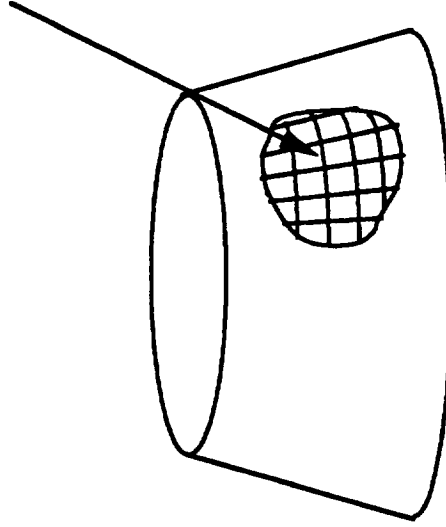
- FLEET BALLISTIC MISSILE (FBM) - SIX GENERATIONS
- LOCKHEED PAYLOAD LAUNCH SYSTEM (LPLS)
- IRIDIUM
- ADVANCED SOLID ROCKET MOTOR (ASRM)
- THEATER HIGH ALTITUDE AREA DEFENSE (THAAD)
- ADVANCED BOMB FAMILY (ABF)
- EXOATMOSPHERIC REENTRY VEHICLE INTERCEPTOR SUBSYSTEM (ERIS)
- NATIONAL LAUNCH SYSTEM (NLS)



VEHICLE DESIGN

STRUCTURAL OPTIMIZATION ON NLS CONICAL INTERSTAGE

INTEGRATED WAFFLE GRID



- OPTIMIZED DESIGN & MATERIAL REDUCED WEIGHT BY 25%
- STUDY ON NLS PAYLOAD CARRIER YIELDED SIMILAR RESULTS
- MANUFACTURING PROCESS EVALUATED:
 - Age forming, shot peening, chem milling



ADVANCED DEVELOPMENT MANUFACTURING

PROFILE

- SELECT CADRE OF DEDICATED, COST CONSCIOUS, MFG PERSONNEL
- EXTENSIVE EXPERIENCE WITH DIVERSE PROTOTYPE PRODUCTS
- INTIMATE KNOWLEDGE OF CONCURRENT ENGINEERING PRACTICES
- ALL CLASSIC MANUFACTURING DISCIPLINES REPRESENTED
- MOCK-UP, PROTOTYPE, & FLIGHT QUALITY HARDWARE
- RESPONSIVE TO ANY LEVEL OF DOCUMENTATION REQUIREMENTS
- INTEGRATED COST/SCHEDULE/MGMT CONTROL SYSTEM
- CONTINUOUS IMPROVEMENT INHERENT TO PERFORMANCE

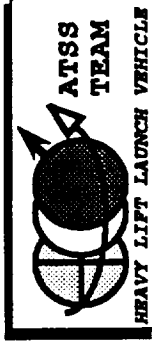


ADVANCED DEVELOPMENT MANUFACTURING

FUNCTIONS

ADM MANUFACTURING SERVICES

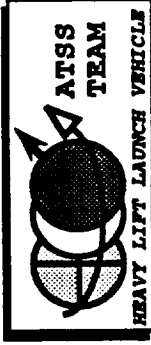
- **MANUFACTURING ENGINEERING**
 - New business support, program/IPDT representation, producibility engineering, cost account management, process planning, tool engineering, electronic/mechanical/industrial engineering support
- **PRODUCTION CONTROL & MFG SUPPORT**
 - Production management, work in process control, cost/schedule mgmt
- **PRODUCT DEVELOPMENT SHOPS**
 - Flight quality hardware, model/mock-up development, CNC/DNC machining capability, clean room operations, welding, composites, adhesive bonding, electronic harnesses, console & box assembly, project tooling, handling & support equipment, precision assembly, sheet metal & weld fabrication, large structures capability



ADVANCED DEVELOPMENT MANUFACTURING

EXPERIENCE SAMPLE

- **NASA**
 - Space Station Freedom, Hubble Space Telescope, Gravity Probe B, Cargo Transfer Vehicle, Assured Crew Return Vehicle, Advanced Solid Rocket Motor
- **NAVY**
 - Fleet Ballistic Missile (FBM) & non FBM products, START covers, Mine Research Systems
- **STRATEGIC DEFENSE/ARMY**
 - ERIS, Starlab, Payload Launch Vehicle/Brilliant Pebbles, Initial National Missile Defense (INMD), Follow-on Early Warning System (FEWS)
- **AIR FORCE**
 - Milstar, Penetrator bomb programs, Advanced Vehicle Experiment (AVEX)



ADVANCED DEVELOPMENT MANUFACTURING

COST EFFICIENT MFG BEGINS WITH EARLY INVOLVEMENT

- GBU-28 "Bunker Buster"
- Personnel Launch System (Feasibility Study)
- Payload Launch Vehicle
- THAAD
- START Covers

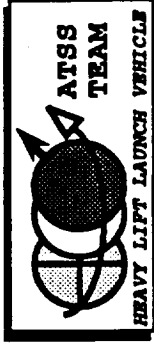
From concept to delivery in 3 1/2 weeks.
First case components in 5 days from concept. Operated under highest government priority "999".
Rapid maturity to viable design concept.
Mfg/program cost understood. NASA pleased at rapid response.
Program Office satisfied customer cost in control. Significant reduction in engineering changes from ERIS.
Mfg & risk mitigated. Confident cost submittal to RFP. Mature tooling concept.
Part count reduction, on cost, on schedule delivery.



ADVANCED DEVELOPMENT MANUFACTURING

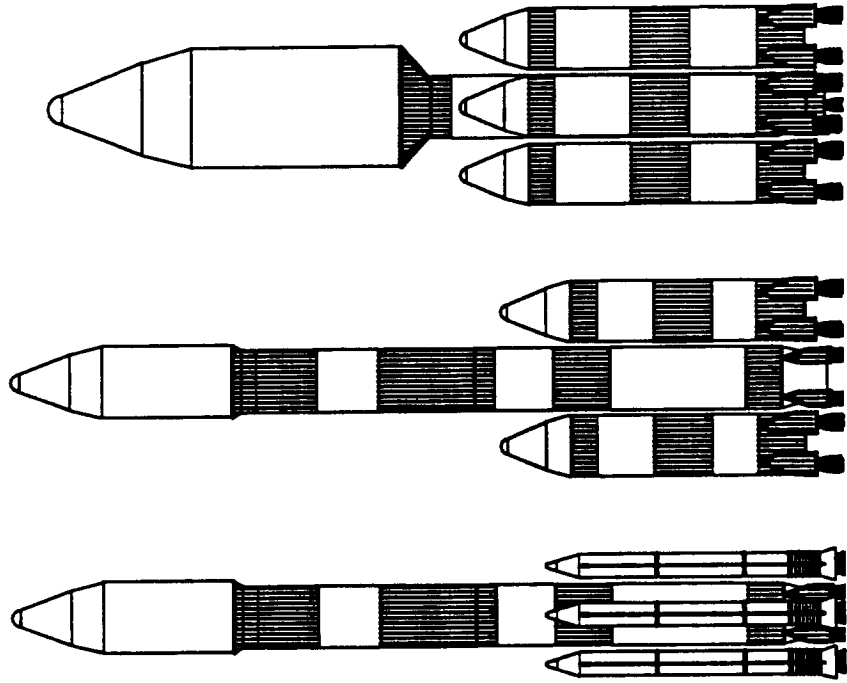
SUMMARY

- ADM PROVIDES THE CUSTOMER WITH:
 - Total manufacturing responsibility
 - Wide range of experience
 - Complete resources
 - Pro-active management
 - Modern management tools
 - Empowered work force
 - "Can do" attitude

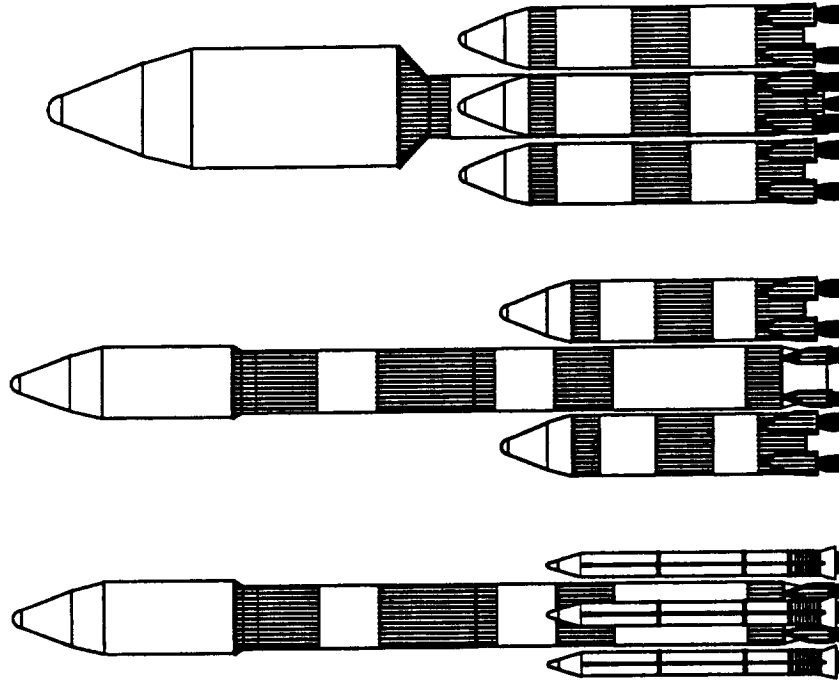


CONCLUSION

- LMSC BRINGS A WIDE VARIETY OF RESOURCES AND TALENT TO THE ATSS PROJECT
- LMSC COMMITTED TO MAKING A SIGNIFICANT CONTRIBUTION TO NASA AND THE ATSS EFFORT



4. FLO Assessment Results To-Date



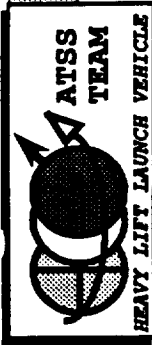
NLS-Derived Common Diameter Concepts

J. McCurry/Lockheed

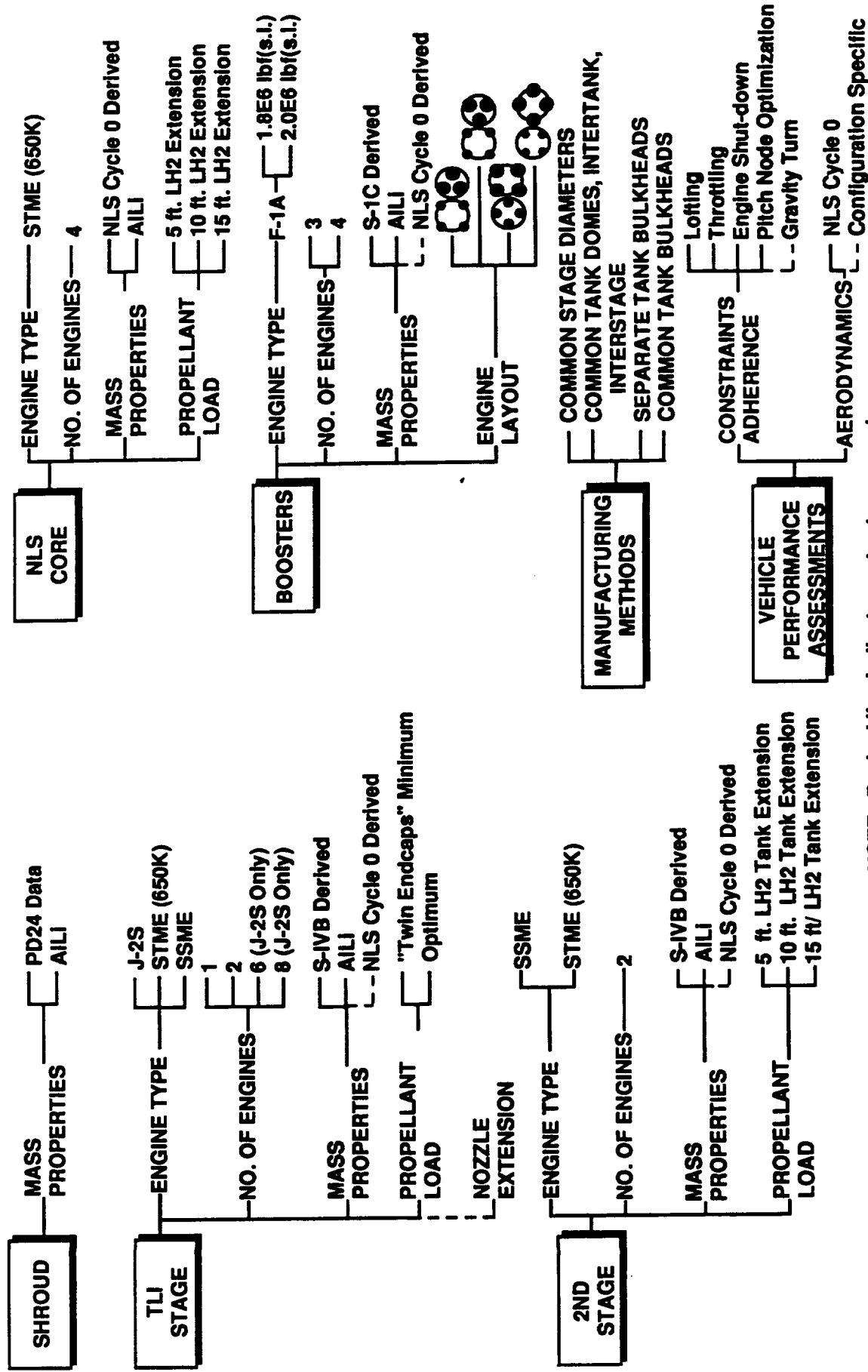


INTRODUCTION

- Lockheed was asked by MSFC/PT01 to develop and assess NLS-derived HLLV concepts that could satisfy the Single Launch lunar mission scenario
- The design goal was to achieve 93 metric tons (or greater) post-TLI (payload plus TLI stage burn-out mass)
- Primary groundrules from PT01 were to have common core and booster tank diameters and F-1As on the boosters
- Rocketdyne was contacted to confirm thrust and throttle sizing options available for the F-1A
 - Up to 2.0E06 lbf sea level thrust
 - Down to 65% RPL minimum throttle level
- Lockheed's design philosophy was to maximize vehicle element commonality, minimize vehicle dry weight, and allow for growth options, while adhering to KSC facility constraints



SINGLE-LAUNCH DESIGN OPTIONS



NOTE: Dashed line Indicates not yet assessed.



GROUNDRULES AND CONSTRAINTS

CORE VEHICLE

- Four STMES
- 5 ft. LH2 tank stretch (1.69E06 lbm propellant load)
- Expendable hardware
- TBD delta weight addition to intertank to carry booster thrust loads
- No air start of core STMES (a manned mission requirement that carries over to unmanned)
- TBD throttle profile; 70% (new STME groundrule) or 100% RPL step throttle



GROUNDRULES AND CONSTRAINTS (Continued)

TLI AND SECOND STAGE

- One or more STMEs, SSMEs, or J-2Ss
- Minimum propellant load limited to dome-to-dome LOX tank design (590,000 lbm total propellant)
- Post-insertion burn or pre-insertion burn (which ever is optimal)
- Mass fraction curves (function of propellant load) for S-IVB derived stage with J-2S, STME, or SSME engines
- 30 day boil-off MPS margin (from STV trade study groundrules)
- Expendable hardware



GROUND RULES AND CONSTRAINTS (Continued)

BOOSTER

- Three or four F-1As per booster
- 75% RPL step throttle (unknown throttle altitude constraint)
- Mass fraction curve (function of propellant load) for S-1C derived booster
- Expendable hardware
- 27.6 ft (NLS-derived) tank diameter
- Boosters support the core vehicle on the pad
- Booster thrust taken out at forward attach



GROUND RULES AND CONSTRAINTS (Continued)

GENERAL

- Flight Performance Reserve (FPR) for each stage is 1% of stage delta V
- 10% inert mass margin for growth
- Unusable propellant is $0.05 \times W_{\text{prop}}$ (from NASA TM 78661, "Techniques for the Determination of Mass Properties of Earth-to-Orbit Transportation Systems", NASA LaRC, June, 1978)
- Lift-off thrust/weight ratio minimum is 1.25:1
- No engine-out protection for making mission
- 4.5 Gs maximum thrust acceleration constraint
- Optimal pitch-rate steering during ascent
- ± 5000 psf-degree Qbar-alpha constraint during atmospheric flight
- Shroud pre-defined by MSFC
- Shroud jettison at 400,000 feet (geodetic altitude)



GROUND RULES AND CONSTRAINTS (Concluded)

OPERATIONS

- 7 days between launches for the Dual Launch scenario
- 60 days (minimum) between launches for the Single Launch scenario
- VAB high bay door height constrains the total vehicle length; limit of 390-400 ft
- VAB high bay crane hook height limit, including height of lifting equipment, imposes a similar vehicle length limit of 390-400 ft
- VAB high bay side door height modifications are determined to be minor in cost impact; current height of 111 ft
- MLP width constrained to current pad support post spacing
- TBD MLP length allowed; limited by crawler overhang

**STACK LIFT-OFF THRUST-TO-WEIGHT RATIO**

F-1As @ 1.8E06 lbf (sea level)	Number of F-1As on a Booster		
	2 F-1As	3 F-1As	4 F-1As
Nominal Core and 2 Boosters	1.051	1.444	1.833
Stretched Core and 2 Boosters	0.951	1.415	1.784
Nominal Core and 4 Boosters	1.099	1.564	2.023
Stretched Core and 4 Boosters	0.989	1.408	1.822

F-1As @ 2.0E06 lbf (sea level)	Number of F-1As on a Booster		
	2 F-1As	3 F-1As	4 F-1As
	1.140	1.558	2.010
	1.032	1.547	1.956
	1.205	1.722	2.231
	1.084	1.550	2.010
Nominal Core and 2 Boosters			
Stretched Core and 2 Boosters			
Nominal Core and 4 Boosters			
Stretched Core and 4 Boosters			

Rule of Thumb: Minimum nominal thrust-to-weight @ lift-off $\geq$ 1.25

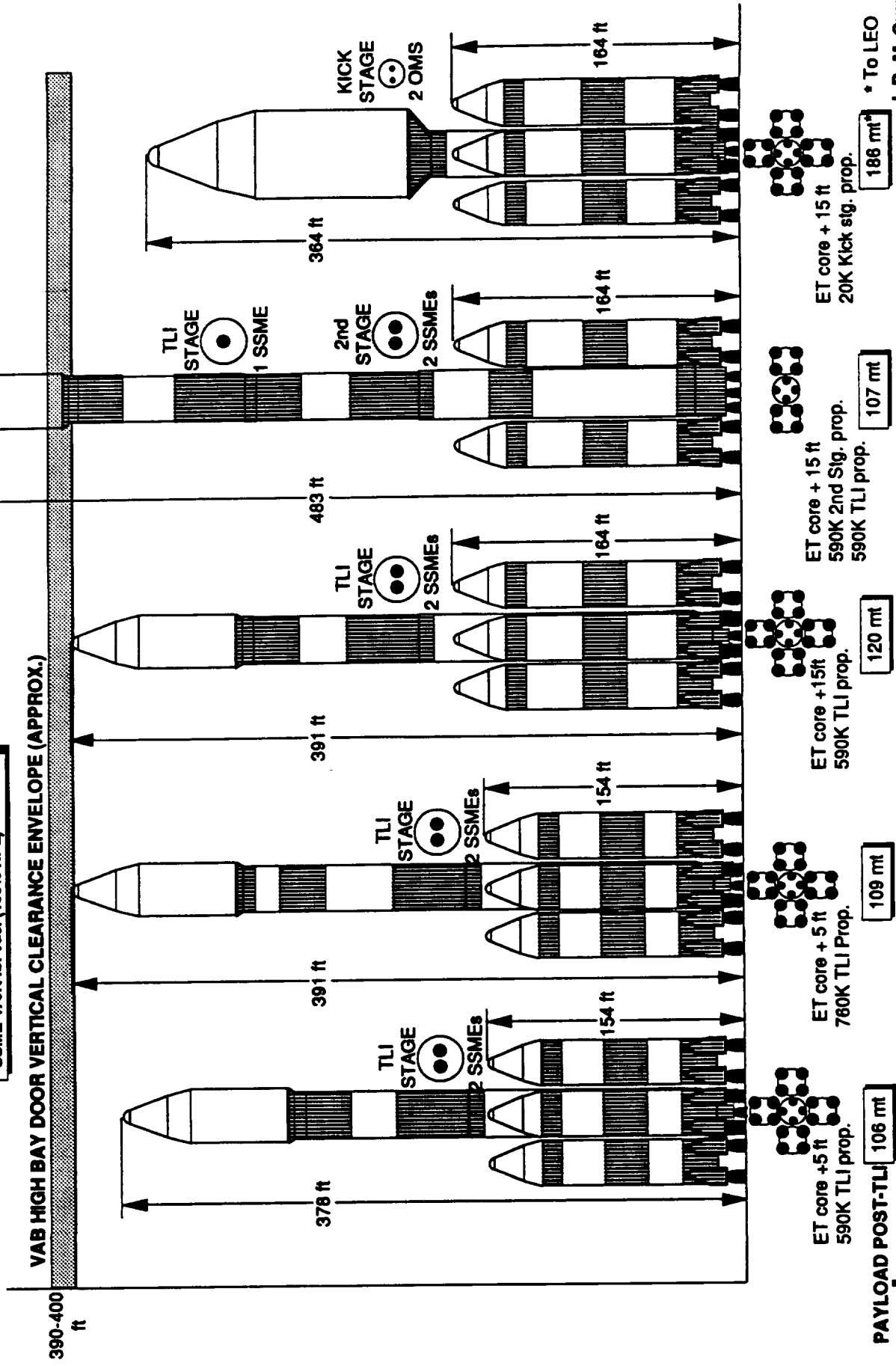
Conclusion: Vehicle configurations with 2 F-1As per booster do not have sufficient thrust to be viable designs.



TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92

SUMMARY OF CANDIDATE HLLV CONFIGURATIONS

For all configurations:
Core: 4 STMEs (650K lbf vac., 100% RPL)
Booster: 4 F-1As (1.8M lbf s.l., 100% RPL)
For upper stages:
SSME 470K lbf vac. (100% RPL)



J. B. McCurry
713-333-6579

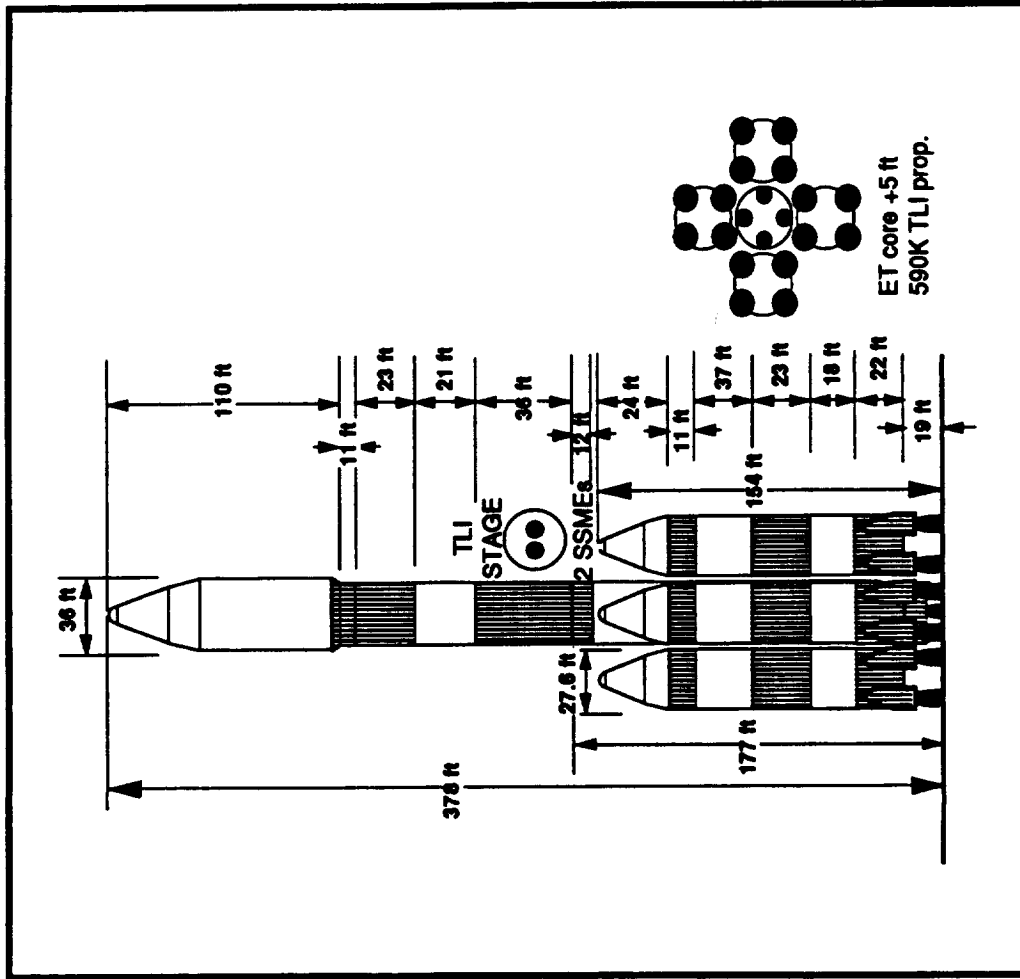




TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92

Lunar Vehicle

Standard NLS Core w/ LOX/RP1 Boosters
and TLI stage (590 k lbf propellant)



Payload: 235 K lbf (106 t)
Final Position: TLI

GLOW: 15,359,000 lbfm

CORE:

Inert Mass: 187.8 K lbfm
Propellant Mass: 1.69 M lbfm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 173 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusability: N.A.

BOOSTER:

Number/Type: 4/ET+ 5 ft
Inert Mass: 235 K lbfm
Propellant Mass: 2.90 M lbfm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 154 ft
Diameter: 27.6 ft
Reusability: N.A.

TLI Stage:

Inert Mass: 78.9 K lbfm
Propellant Mass: 0.59 M lbfm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusability: N.A.

SHROUD - Usable Volume: 33 x60 ft; Mass: 35,500 lb
Comments: • ET prop. capacity based on a 5 ft stretch
• STME has 75% step throttle
• Max. G = 4.5 Max. Q = 900 psf



J. B. McCurry
713-333-6579

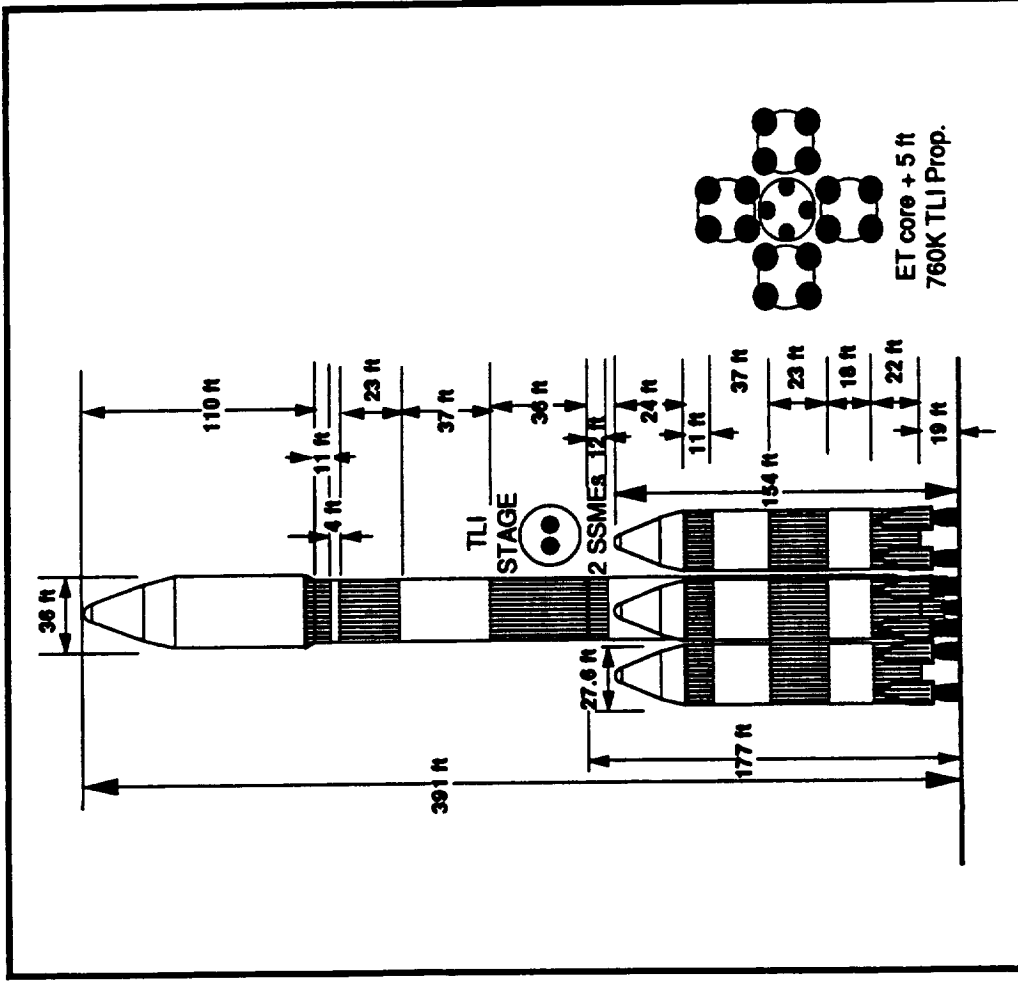


Lunar Vehicle

Standard NLS Core w/ LOX/RP1 Boosters
and TLI stage (760 k lbf propellant)

Payload: Final Position:	240 K lbf (109 t) TLI
---	--

GLOW: 15,551,000 lbfm



SHROUD - Usable Volume: 33 x60 ft; Mass: 35,500 lb
Comments: • ET prop. capacity based on a 5 ft stretch
• STME has 75% step throttle
• Max. G = 4.5 Max. Q = 900 psf

CORE:

Inert Mass: 187.8 K lbfm
Propellant Mass: 1.69 M lbfm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 173 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusability: N.A.

BOOSTER:

Number/Type: 4/ET+ 5 ft
Inert Mass: 235 K lbfm
Propellant Mass: 2.90 M lbfm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 154 ft
Diameter: 27.6 ft
Reusability: N.A.

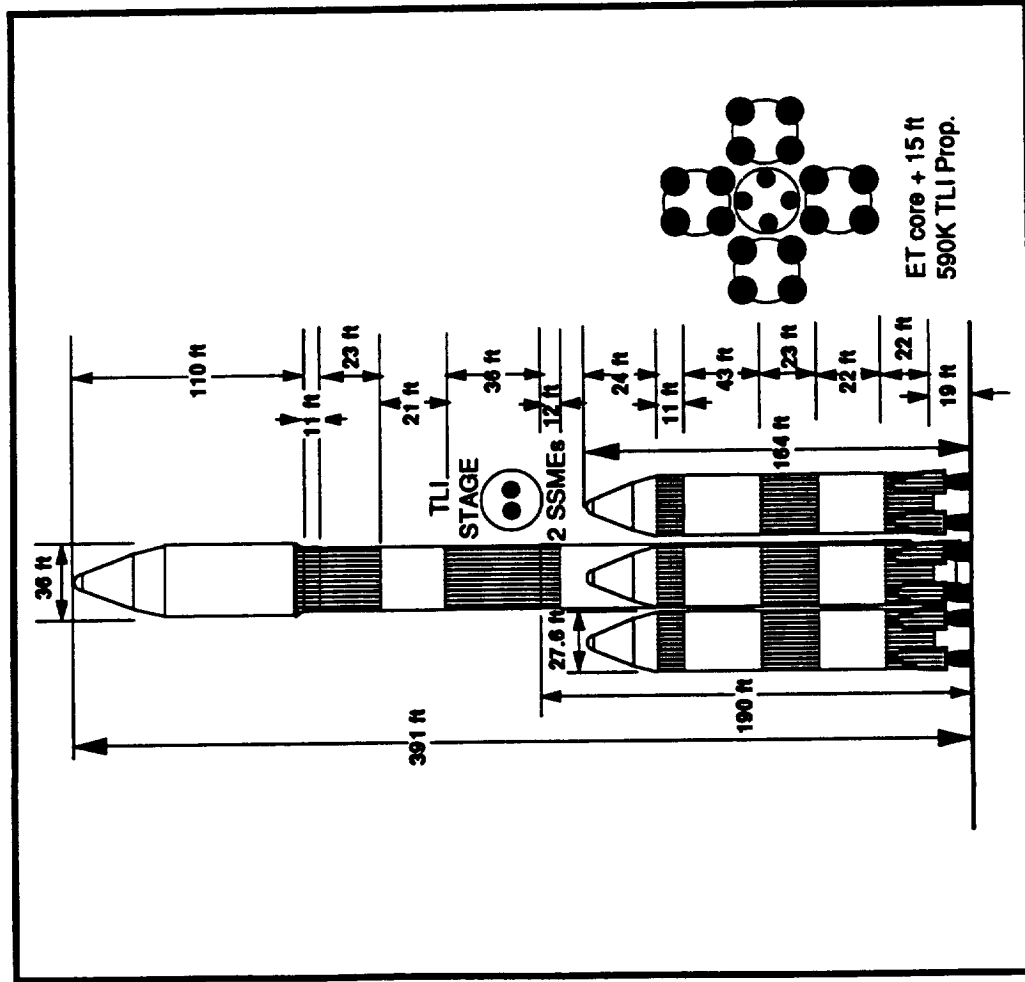
TLI Stage:

Inert Mass: 95.9 K lbfm
Propellant Mass: 0.76 M lbfm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5sec
Engine Exit Dia.: 96 in.
Length: 101 ft
Diameter: 27.6 ft
Reusability: N.A.



Lunar Vehicle

Stretched NLS Core w/ LOX/RP1 Boosters
and TLI stage (590 k lbm propellant)



SHROUD - Usable Volume: 33 x 60 ft; Mass: 35,500 lb
Comments: • ET prop. capacity based on a 15 ft stretch
• STME has 75% step throttle
• Max. G = 4.5 Max. Q = 900 psf

Payload: 265 K lbm (120 t)
Final Position: TLI

GLOW: 17,077,000 lbm

CORE:

Inert Mass: 206.4 K lbm
Propellant Mass: 1.86 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 186 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusability: N.A.

BOOSTER:

Number/Type: 4/ET + 15 ft
Inert Mass: 251 K lbm
Propellant Mass: 3.26 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 164 ft
Diameter: 27.6 ft
Reusability: N.A.

TLI Stage:

Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusability: N.A.

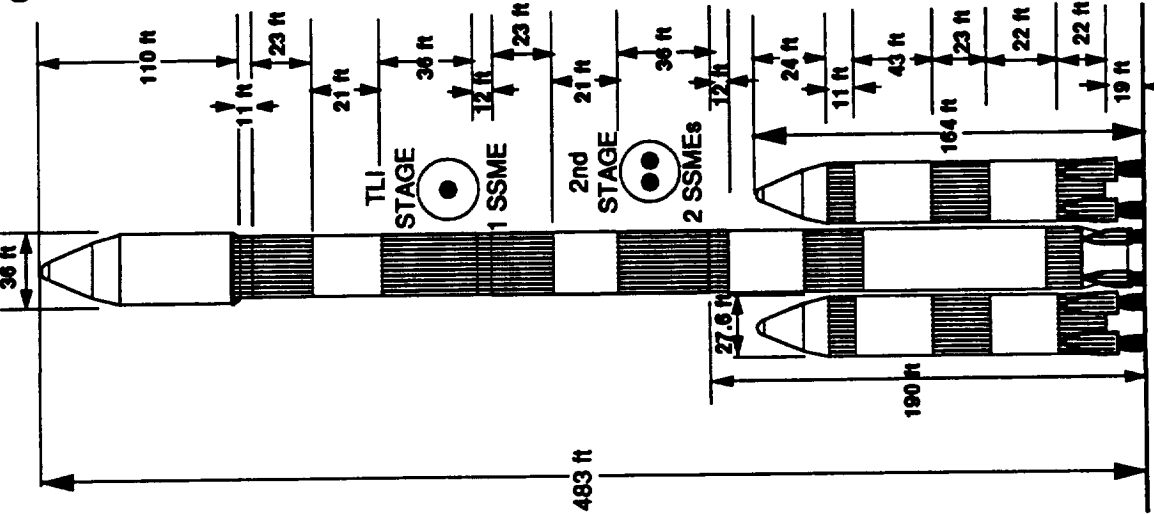


Lunar Vehicle

Stretched NLS Core w/ LOX/RP1 Boosters, 2nd Stage, (590 k lbm propellant) and TLI stage (590 klbm propellant)

Comments:

- ET prop. capacity based on a 15 ft stretch
- STME has 75% step throttle
- Max. $G = 4.5$ Max. $Q = 900$ psf
- SHROUD - Usable Volume: 33 x60 ft; Mass: 35,500 lbm



Payload: 236 K lbm (107 t)
Final Position: TLI

GLOW: 10,686,000 lbm

CORE:

Inert Mass: 206.4 K lbm
Propellant Mass: 1.86 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 186 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusability: N.A.

BOOSTER:

Number/Type: 2/ET+ 15 ft
Inert Mass: 251 K lbm
Propellant Mass: 3.26 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 164 ft
Diameter: 27.6 ft
Reusability: N.A.

TLI Stage:

Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/1
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusability: N.A.



TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92

Mars Vehicle

Stretched NLS Core w/ LOX/RP1 Boosters
and Kick Stage (14.3 K lbm propellant)

Payload: 410 K lbm (186 t)
Final Position: 220 nm LEO

GLOW: 16,696,000 lbm

CORE:

Inert Mass: 206.4 K lbm
Propellant Mass: 1.86 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 186 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:

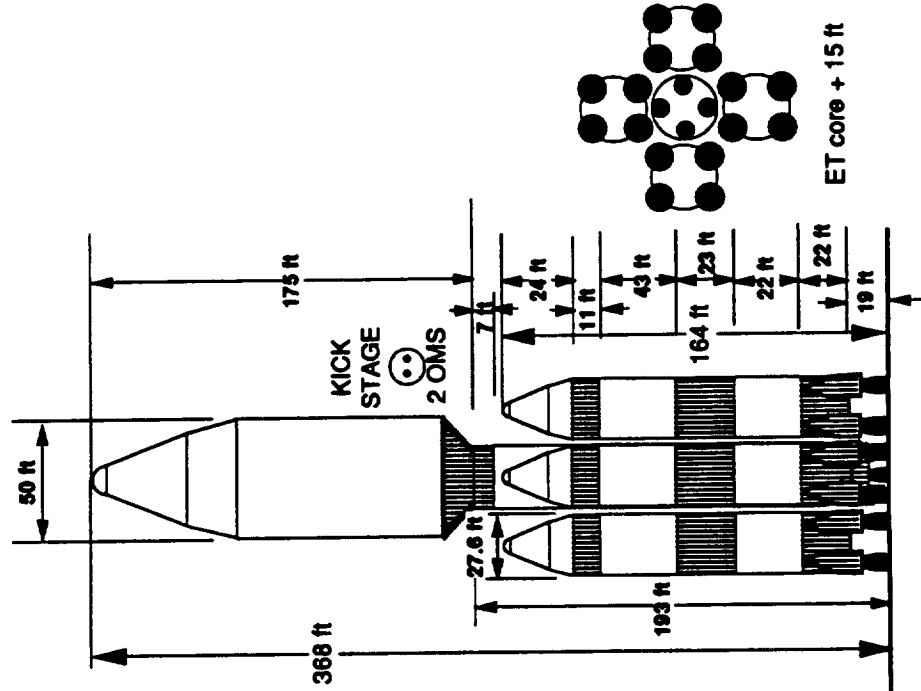
Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusability: N.A.

BOOSTER:

Number/Type: 4/ET+ 15 ft
Inert Mass: 251 K lbm
Propellant Mass: 3.26 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 164 ft
Diameter: 27.6 ft
Reusability: N.A.

KICK STAGE:

Inert Mass: 3.6 K lbm
Propellant Mass: 14.3 K lbm
Propellant Type: NTO/MMH
Engine Type/No.: OMS/2
Vac Thrust (ea.): 6.0 K lbf
Vac ISP: 313.0 sec
Engine Exit Dia.: 21.5 in.
Length: 6 ft
Diameter: 27.6 ft
Reusability: N.A.



SHROUD - Usable Volume: 50 x 100 ft; Mass: 160,000 lb

Comments: • ET prop. capacity based on a 15 ft stretch

• STME has 75% step throttle

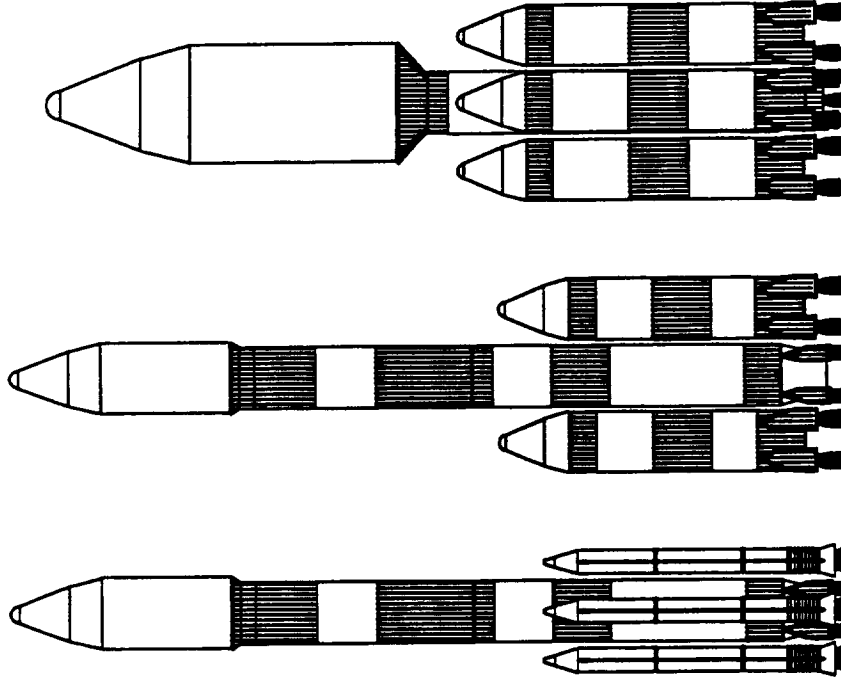
• Max. G = 4.5 Max. Q = 900 psf





CONCLUSIONS

- To meet a lunar mission requirement of >90 mt post-TLI and not violate the VAB high bay door clearance, must use four F-1A powered boosters with the NLS-derived core
- Use of four F-1As on four NLS-derived boosters with 2 SSMEs on the TLI stage provide approximately 115 mt of payload post-TLI
- Use of F-1A throttling for Qbar control gains approximately 30,000 lbm of payload post-TLI
- Use of STMEs for upper stage applications cannot compete with SSMEs from vehicle sizing and performance standpoints
- Use of 2 boosters with 3 F-1As (1.8E6 lbf s.l. thrust) will not allow greater than 90 mt post-TLI, without using both a second stage and a TLI stage
- Use of common propellant tank, intertank, aft skirt, and forward skirt/interstage piece-parts still allows reasonable vehicle configurations to be designed, without incurring inordinate performance losses
 - Associated cost savings could offset non-optimality of the design
- The VAB high bay doors can be economically modified, up to a point, to accommodate booster height, but not economically for core vehicle height



Monolithic Concepts

K. Holden/Lockheed



INTRODUCTION

- Lockheed was asked to identify and assess candidate monolithic heavy lift launch vehicles for the first lunar outpost and mars missions
 - Common diameter for all stages
 - 38 foot (lunar mission payload shroud diameter)
 - 50 foot (mars mission payload shroud diameter)
- Several first order design parameters were assessed
 - Number of stages
 - Stage propellant combination
 - Type of stage engine
 - Number of stage engines
 - Stage tankage configurations
- All candidate configurations were series-burn vehicles with no strap-on boosters
- Two vehicle sizing iterations were performed
 - Initial design
 - Alternate vehicle tankage options



INITIAL DESIGN ITERATION

This initial analysis looked at the following options:

- Payload to Trans-lunar injection (TLI)
- Two, three and four stage vehicle were analyzed
- Both RP-1 and LH2 were used as second stage fuels
- One and two SSMEs were used on TLI the stage
- STMEs were substituted for SSMEs on the second stage
- STMEs were substituted for SSMEs on the second and TLI stages



INITIAL DESIGN ITERATION (Concluded)

Groundrules:

- Figure of merit was number of engines
- Two, three and four stage vehicles were considered
- First stage engines were F-1As
- Second stage engines were F-1As, SSMEs or STMEs
- Third and fourth stage engines were SSMEs and STMEs
- Saturn stage mass fractions, interstage weights and instrument unit weights were used
- Payload was 200,000 lbm (90,700 kg) to TLI
- Rocket equation program was used to do sizing
- Mission velocity to TLI varied as a function of initial thrust-to-weight
 - If No1= 1.40 g, mission velocity= 40,700 ft/sec
 - If No1= 1.50 g, mission velocity= 40,200 ft/sec



INITIAL DESIGN ITERATION CONFIGURATION MATRIX

POSSIBLE MONOLITHIC SOLUTIONS

CASE	STAGE 1			STAGE 2			STAGE 3			STAGE 4		
	ENGINE	No1	W11 (lbm)	ENGINE	No2	W12 (lbm)	ENGINE	No3	W13 (lbm)	ENGINE	No4	W14 (lbm)
1	12 F-1A	1.511	11,000,000	5 SSME	0.981	2,000,000	-	-	-	-	-	-
2	10 F-1A	1.470	7,200,000	2 F-1A	0.908	2,300,000	3 SSME	0.721	1,600,000	-	-	-
3	7 F-1A	1.416	6,000,000	5 SSME	0.978	1,200,000	2 SSME	0.870	800,000	-	-	-
4	7 F-1A	1.414	6,000,000	6 SSME	1.168	1,500,000	1 SSME	0.668	450,000	-	-	-
5	7 F-1A	1.456	4,800,000	5 STME	0.941	2,500,000	1 SSME	0.663	450,000	-	-	-
6	7 F-1A	1.430	4,600,000	5 STME	0.849	2,650,000	1 STME	0.707	650,000	-	-	-
7	6 F-1A	1.466	4,200,000	6 SSME	1.002	1,500,000	2 SSME	0.807	600,000	1 SSME	0.969	250,000

- Two Stage to TLI is not competitive with three stage to TLI
- There was no advantage in going to a four stage vehicle



ALTERNATE VEHICLE TANKAGE OPTIONS

- Case four was selected for detailed analysis
- Programs were written to estimate stage weights for three vehicle configurations
 - Configuration A
 - 38 foot stage diameter
 - Stages one and two used propellant tanks with elliptical end caps
 - Stage three used a cluster of seven propellant tanks (two for LOX and five for LH2)
 - Configuration B
 - 38 foot stage diameter
 - Lower propellant tank on all stages used toroidal end caps
 - Configuration C
 - 50 foot stage diameter
 - Lower propellant tanks on stages one and two used toroidal end caps
 - Stage three used a cluster of seven propellant tanks (two for LOX and five for LH2)



ALTERNATE VEHICLE TANKAGE OPTIONS (Continued)

- Weight estimating algorithm based on "TECHNIQUES FOR THE DETERMINATION OF MASS PROPERTIES OF EARTH-TO-ORBIT TRANSPORTATION SYSTEMS", NASA TM 78661, by I. O. MacConochir and P. J. Klich, June 1976
- Additional data on avionics and thrust vector control (TVC) weights from "NLS DESIGN TEAM WEIGHT STATUS REPORT#2 CYCLE 0 CONFIGURATION ", FEBRUARY, 1992
- Additional data on TLI stage thermal protection system (TPS) and stage separation/ullage motors from MSFC in house "heavy Lift Launch Vehicle Definition", Status report, April 30, 1992
- Unpressurized structure weights from "The Influence of Unconventional Structures and Advanced Materials on Booster Reusability", by Philip Bono, Fifth Annual AIAA Structures and Materials Conference, Palm Springs CA, April 1-3, 1964
- Engine data from Rocketdyne brochures received from T. Murphy/Rocketdyne
- Propellant tank weights from "A Semi-empirical Method for Propellant Tank Weight Estimation", by L. Willoughby, 27 th Society of Aeronautical Weight Engineers Annual Conference, New Orleans, LA, May 13-16, 1968



ALTERNATE VEHICLE TANKAGE OPTIONS (Continued)

- Cases 4A, 4B and 4C were sized by a rocket equation program to have a payload of 205,000 lbm post TLI
- Case 4A
 - Stage 1, seven F-1As, propellant load= 5,700,000 lbm, propellant mass fraction (PMF)= 0.92957, No1= 1.441 Gs
 - Stage 2, six SSMEs, propellant load= 1,650,000 lbm, PMF= 0.90347, No2= 1.092 Gs
 - Stage 3, one SSME, propellant load= 500,000 lbm, PMF= 0.90603, No3= 0.621 Gs
 - Total vehicle length= 388 feet
 - Stage 2 was constrained by minimum size of LOX tank



ALTERNATE VEHICLE TANKAGE OPTIONS (Continued)

- Case 4 sizing (Continued)
 - Case 4B
 - Stage 1, seven F-1As, propellant load= 5,700,000 lbm, PMF= 0.93069, No1= 1.444 Gs
 - Stage 2, six SSMEs, propellant load= 1,550,000 lbm, PMF= 0.90296, No2= 1.095 Gs
 - Stage 3, one SSME, propellant load= 600,000 lbm, PMF= 0.91679, No3= 0.547 Gs
 - Total vehicle length= 377 feet
 - Stage 3 was constrained by minimum size of LH2 tank



ALTERNATE VEHICLE TANKAGE OPTIONS (Continued)

- Case 4 sizing (Concluded)
 - Case 4C
 - Stage 1, seven F-1As, propellant load= 5,700,000 lbm, PMF= 0.92969 No1= 1.442 Gs
 - Stage 2, six SSMEs, propellant load= 1,650,000 lbm, PMF= 0.90314, No2= 1.093 Gs
 - Stage 3, one SSME, propellant load= 500,000 lbm, PMF= 0.91145, No3= 0.624 Gs
 - Total vehicle length= 321 feet

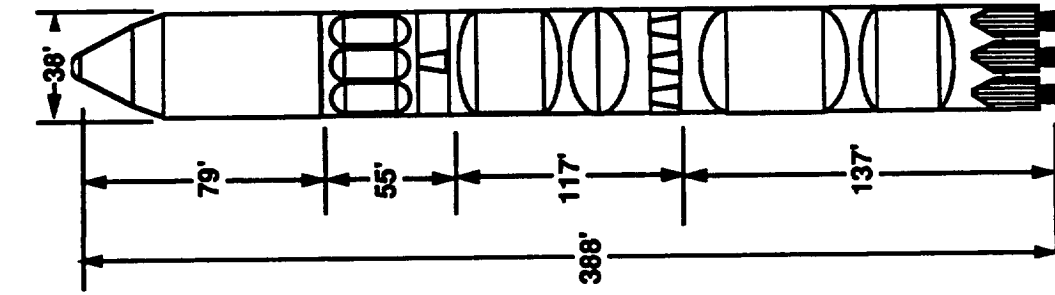


ALTERNATE VEHICLE TANKAGE OPTIONS (Concluded)

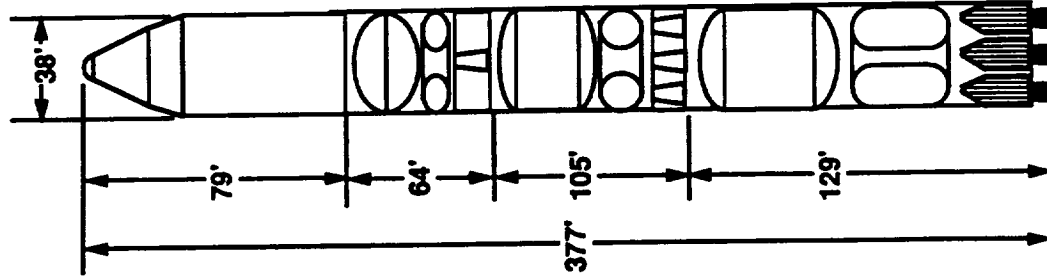
- Performance (payload to TLI) was verified by SORT trajectory analysis
 - Assumptions
 - Maximum acceleration= 4.0 g
 - Max Q= 900 psf
 - Max Q-alpha= 5000 psf-deg
 - Launch azimuth= 72 deg
 - Payload
 - Case 4A= 209,556 lbm
 - Case 4B= 212,440 lbm
 - Case 4C= 212,098 lbm



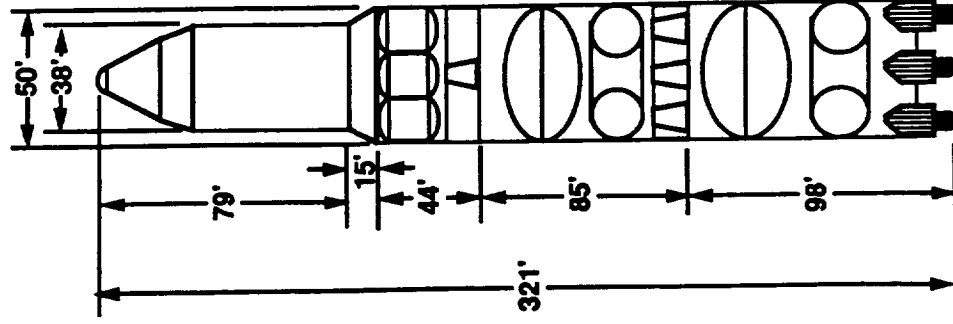
CASE 4 ALTERNATIVE DESIGNS



Case 4A
Wpay = 209,556 lbm
(95 mt)



Case 4B
Wpay = 212,440 lbm
(96 mt)



Case 4C
Wpay = 212,690 lbm
(96 mt)



DEGRADED THIRD STAGE PERFORMANCE

- The following data were the results of an analysis of the performance of the Case 4A vehicle with degraded third stage PMF
 - Case 4A (Nominal third stage PMF)
 - Stage 1, seven F-1As, propellant load= 5,700,000 lbm, propellant mass fraction (PMF)= 0.92957, No1= 1.441 Gs
 - Stage 2, six SSMEs, propellant load= 1,650,000 lbm, PMF= 0.90347, No2= 1.092 Gs
 - Stage 3, one SSME, propellant load= 500,000 lbm, PMF= 0.90603, No3= 0.621 Gs
 - Total vehicle length= 388 feet
 - Stage 2 was constrained by minimum size of LOX tank
 - Payload= 209,556 lbm



DEGRADED THIRD STAGE PERFORMANCE

- Case 4 results (Continued)
 - Case 4A2 (Third stage PMF= 0.89)
 - Stage 1, seven F-1As, propellant load= 5,500,000 lbm, PMF= 0.92762, No1= 1.441 Gs
 - Stage 2, six SSMEs, propellant load= 2,000,000 lbm, PMF= 0.91324, No2= 1.011 Gs
 - Stage 3, one SSME, propellant load= 350,000 lbm, PMF= 0.89, No3= 0.786 Gs
 - Total vehicle length= 393 feet
 - Payload= 203,098 lbm

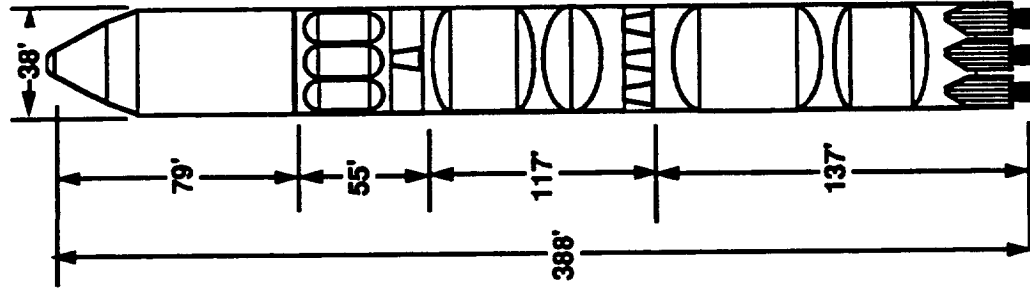


DEGRADED THIRD STAGE PERFORMANCE

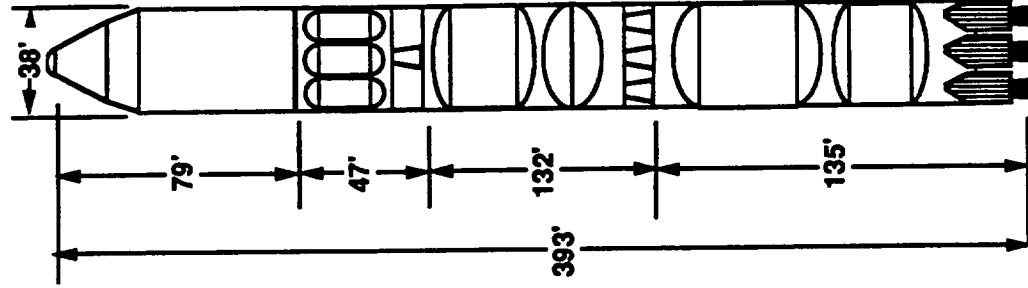
- Case 4 results (Concluded)
 - Case 4A3 (Third stage PMF= 0.88)
 - Stage 1, seven F-1As, propellant load= 5,300,000 lbm, PMF= 0.92553, No1= 1.448 Gs
 - Stage 2, six SSMEs, propellant load= 2,150,000 lbm, PMF= 0.91647, No2= 0.956 Gs
 - Stage 3, one SSME, propellant load= 350,000 lbm, PMF= 0.88, No3= 0.780 Gs
 - Total vehicle length= 396 feet
 - Payload= 207,802 lbm



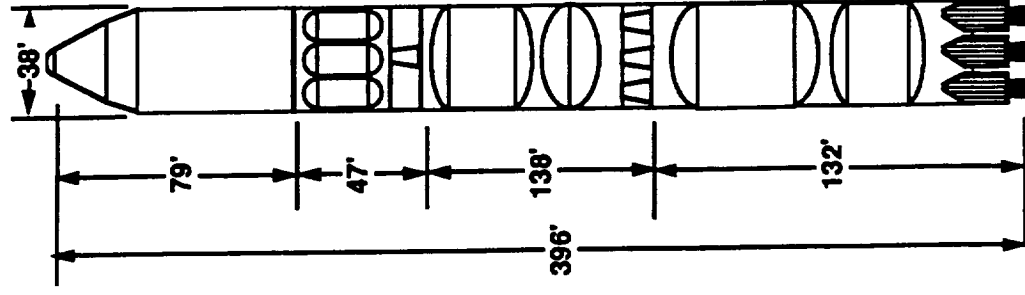
DEGRADED THIRD STAGE PERFORMANCE (Concluded)



Case 4A, PMF3= 0.90603



Case 4A2, PMF3= 0.89

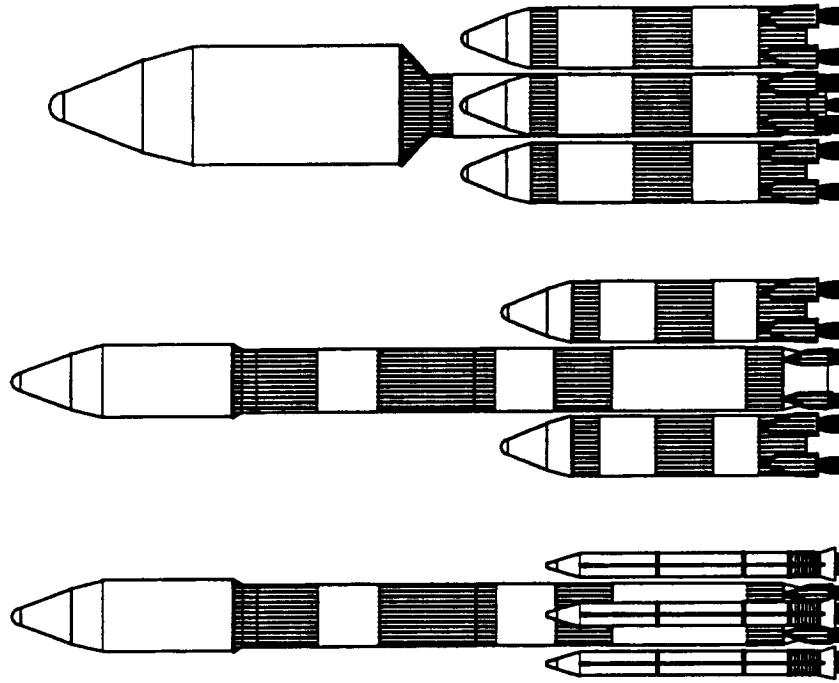


Case 4A3, PMF3= 0.88



THINGS TO DO NEXT

- Additional tankage designs need to be considered
- Research should be done on propellant residual weights as a function of tankage design
- Analysis should be done on cases 3, 5 and 6



Ground Operations Assessments

G. Letchworth/Lockheed



TA-2 HLLV Development Contract Kick-Off Meeting; 6/17/92

First Lunar Outpost Activities/Assessments

LSOC Support to KSC PT (March - May 1992)

- Alternate Launch Site Assessment– June
- SEI Launch Site Operations (Moon/Mars Red Team)– June
- Launch Processing Scenarios and Timelines– March thru May
 - Single and Dual Launch Concepts
 - Saturn vs. NLS-Derived Core
 - Saturn vs. NLS-Derived Boosters
- Recovery Element Assessment– April
- Launch Site Facility Assessment– March (Internal Study)





NASA
Kennedy Space Center

JUNE 16, 1992

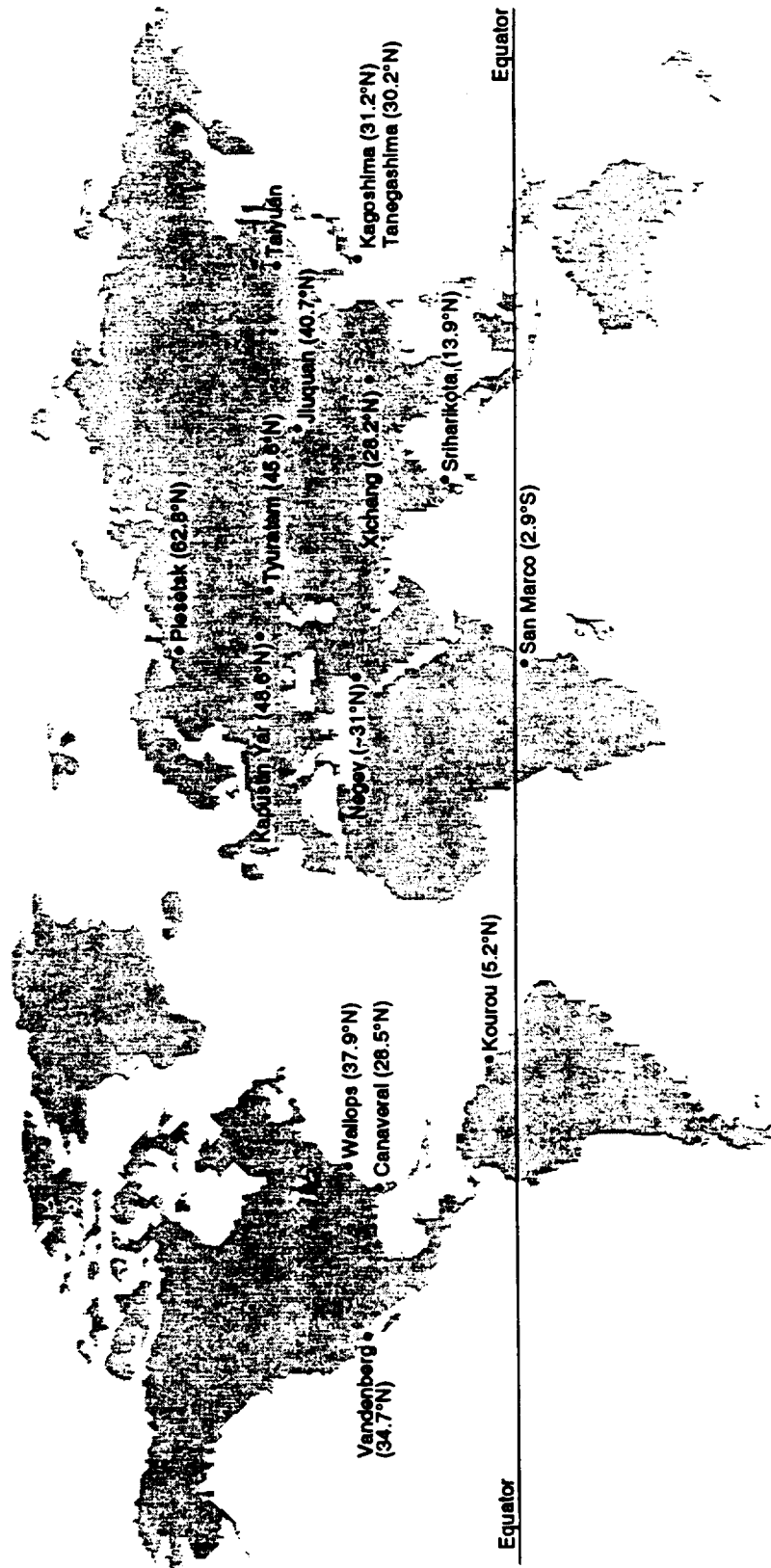
Space Exploration Initiative (SEI)

ALTERNATE LAUNCH SITES ASSESSMENT

JOEL S. BLUM
KSC-PT



EXISTING LAUNCH SITE CANDIDATES



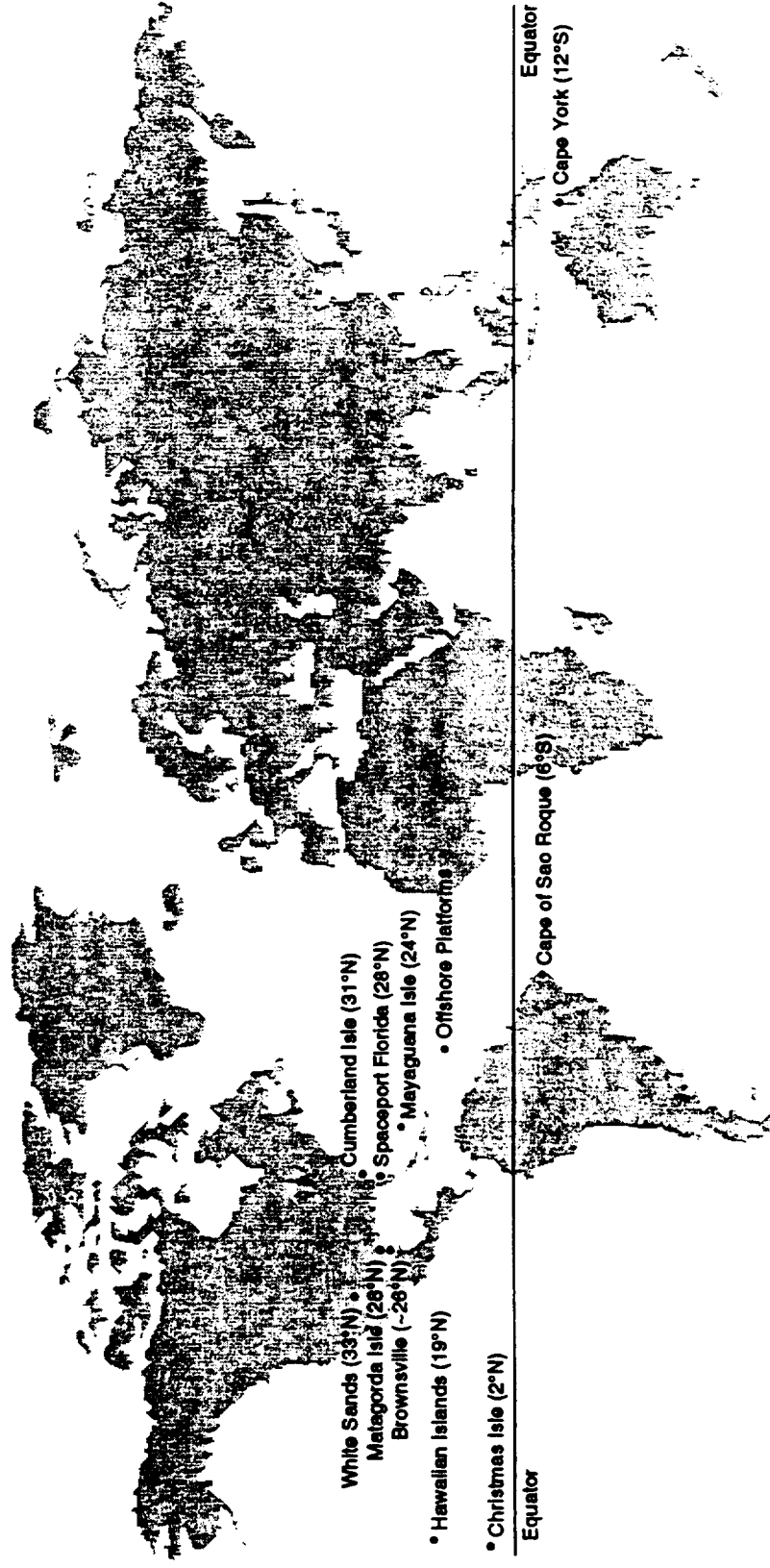


EXISTING LAUNCH SITE CANDIDATES

Cape Canaveral	USA	28.5°N	Delta II/ Atlas I; II; IIA; IIAS/ Titan III; IV
Jiuquan	PRC	40.7°N	Long March-1D; 2C
Kagoshima	Japan	31.2°N	M-3S-II; M-V
Kapustin Yar	CIS	48.4°N	Kosmos
Kennedy Space Center	USA	28.6°N	Space Shuttle
Kourou	French Guiana	5.2°N	Ariane 40; 42P; 44P; 44LP; 44L; 5; 5/Hermes
Negev	Israel	31.0°N	Shavit
Plesetsk	CIS	62.8°N	Kosmos/ Tsyklon/ Molniya/ Soyuz
San Marco	Kenya	2.9°S	Scout/ Enhanced Scout
Sriharikota	India	13.9°N	SLV-3/ ASLV/ PSLV/ GSLV
Taiyuan	PRC	TBD	Long March-4
Tanegashima	Japan	30.2°N	H-1; H-2
Tyuratam	CIS	45.6°N	F-1-M/ Vostok/ Molniya/ Souyz/ Zenit-2; 3/ Proton/ Energia/ Buran
Vandenberg	USA	34.7°N	Scout/ Enhanced Scout/ Taurus/ Delta II/ Atlas E/ Titan II SLV; IV
Wallops	USA	37.9°N	Scout/ Enhanced Scout
Xichang	PRC	28.2°N	Long March-3; 3A; 2E; 2E/HO



POTENTIAL NEW LAUNCH SITES





Space Exploration Initiative

ALTERNATE LAUNCH SITES ASSESSMENT

TOP LEVEL LAUNCH SITE SELECTION FIGURES OF MERIT

COST

Land Acquisition
Land Reclamation
Environmental
Infrastructure
Construction of
Facilities
Equipment
Activation
Operations
Maintenance
Logistics

SCHEDULE

Land Acquisition
Land Reclamation
Environmental
Infrastructure
Development
Construction of
Facilities
Equipment
Activation

PERFORMANCE

Latitude
Altitude
Temperature
Atmospheric
Density

SUPPORTABILITY

Operations
Maintenance
Logistics



GROUND SYSTEM DESIGN CANDIDATE SELECTION CRITERIA

■ LOGISTICS

- *RECOVERY SUPPORT FOR RECOVERABLE HARDWARE*
- *COMMODITIES DELIVERY COSTS (MAKE OR BUY)*
- *FLIGHT HARDWARE DELIVERY COSTS*
- *PAYLOAD/CARGO DELIVERY COSTS*
- *CONSTRUCTION MATERIALS DELIVERY COSTS*
- *REMOTE SITE SUPPORT COSTS*
- *OPERATIONS AND MAINTENANCE COSTS*

■ DUPLICATION OF EXISTING LAUNCH CAPABILITIES

- *FACILITIES*
- *EQUIPMENT*
- *PERSONNEL*
- *SERVICES*



Space Exploration Initiative

ALTERNATE LAUNCH SITES ASSESSMENT

FLIGHT DESIGN CANDIDATE SELECTION CRITERIA

- **LATITUDE VS. ORBITAL INCLINATION**
 - 1525 FT/SEC VELOCITY AT EQUATOR
 - 1340 FT/SEC VELOCITY AT KSC/CCAFS
- **LAUNCH AZIMUTH CONSTRAINTS**
 - NEIGHBORING LAND MASSES
 - RANGE SAFETY (LAUNCH, REENTRY DEBRIS FOOTPRINTS, SUB-ORBITAL DISPOSAL)
- **ALTITUDE OF LAUNCH PAD**
- **LAUNCH SITE AVERAGE WEATHER AND WINDSPEEDS**

CONT'D



NASA
Kennedy Space Center

JUNE 3, 1992

Space Exploration Initiative (SEI)

SEI LAUNCH SITE OPERATIONS AND FACILITIES

KSC PRESENTATION TO MOON/MARS RED TEAM



Space Exploration Initiative

LAUNCH SITE OPERATIONS AND FACILITIES

KSC PROCESSING ASSUMPTIONS

FASTER, BETTER, CHEAPER:

- FLIGHT HARDWARE IS DESIGNED FOR OPERABILITY TO REDUCE MANPOWER AND GROUND PROCESSING TIMELINES (FLIGHT ELEMENTS UTILIZE "SHIP AND SHOOT" APPROACH)

- FLIGHT HARDWARE ARRIVING AT THE LAUNCH SITE IS READY FOR FLIGHT:

- MAJOR ELEMENTS FULLY ASSEMBLED
- NO OPEN PAPER
- NO DEFERRED WORK
- NO MODIFICATIONS



Space Exploration Initiative

LAUNCH SITE OPERATIONS AND FACILITIES

KSC PROCESSING ASSUMPTIONS

FASTER, BETTER, CHEAPER:

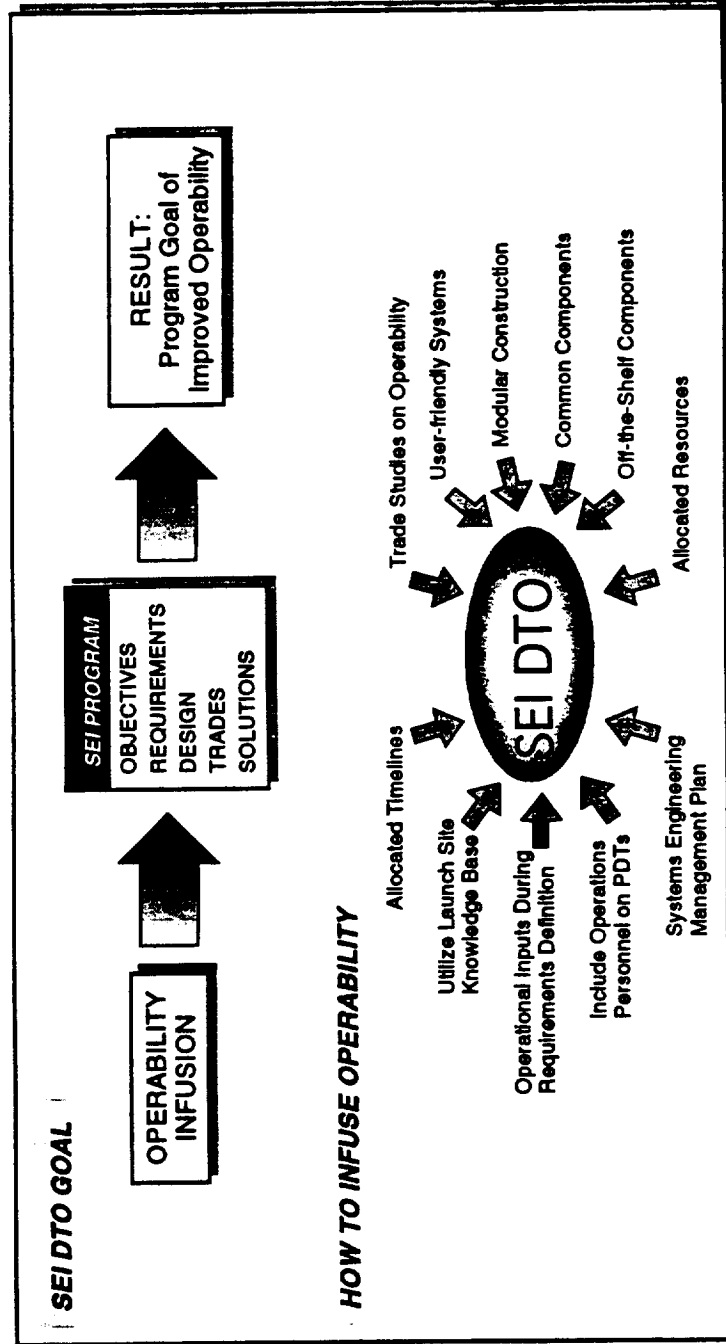
- LAUNCH SITE TEST AND CHECKOUT REQUIREMENTS ARE SIGNIFICANTLY REDUCED:
 - MINIMAL ELEMENT / COMPONENT TESTING; EMPHASIS ON INTEGRATED TESTING
 - UTILIZE VEHICLE HEALTH MONITORING
 - DO NOT REPEAT FACTORY TESTING
- FLIGHT HARDWARE ACCESS AT THE LAUNCH PAD IS VERY LIMITED
- PAPER SYSTEM UTILIZED FOR FLIGHT HARDWARE PROCESSING IS NOT LABOR INTENSIVE



Space Exploration Initiative

LAUNCH SITE OPERATIONS AND FACILITIES

DESIGN-TO-OPERATIONS (DTO)



SEI DTO GOALS AND METHODS REQUIRED TO ACHIEVE THEM

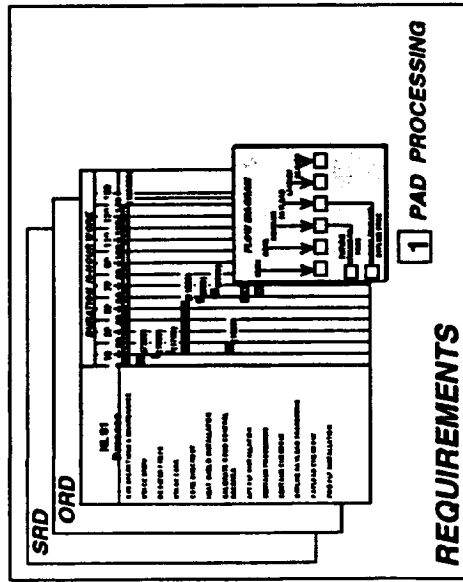


Space Exploration Initiative

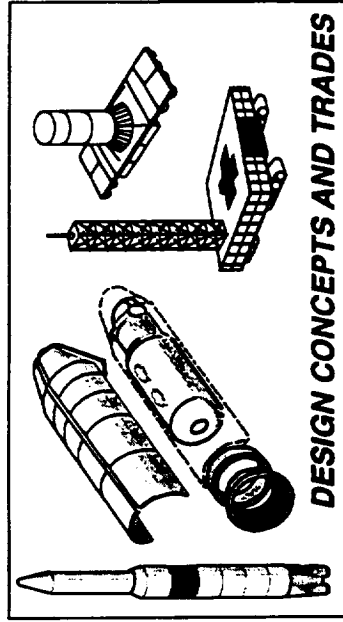
LAUNCH SITE OPERATIONS AND FACILITIES

ALLOCATED AND ASSESSED TIMELINE COMPARISON

ALLOCATED TIMELINES

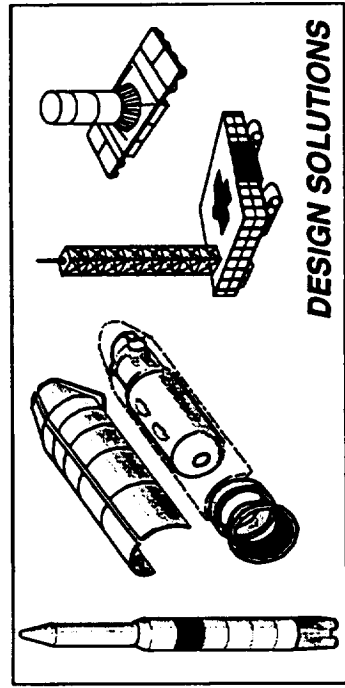


INCORPORATION

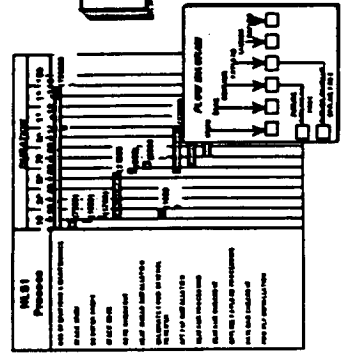


RE-ASSESSMENT

ASSESSED TIMELINES



ANALYSIS



**OPTIMUM
DESIGN
SOLUTION**

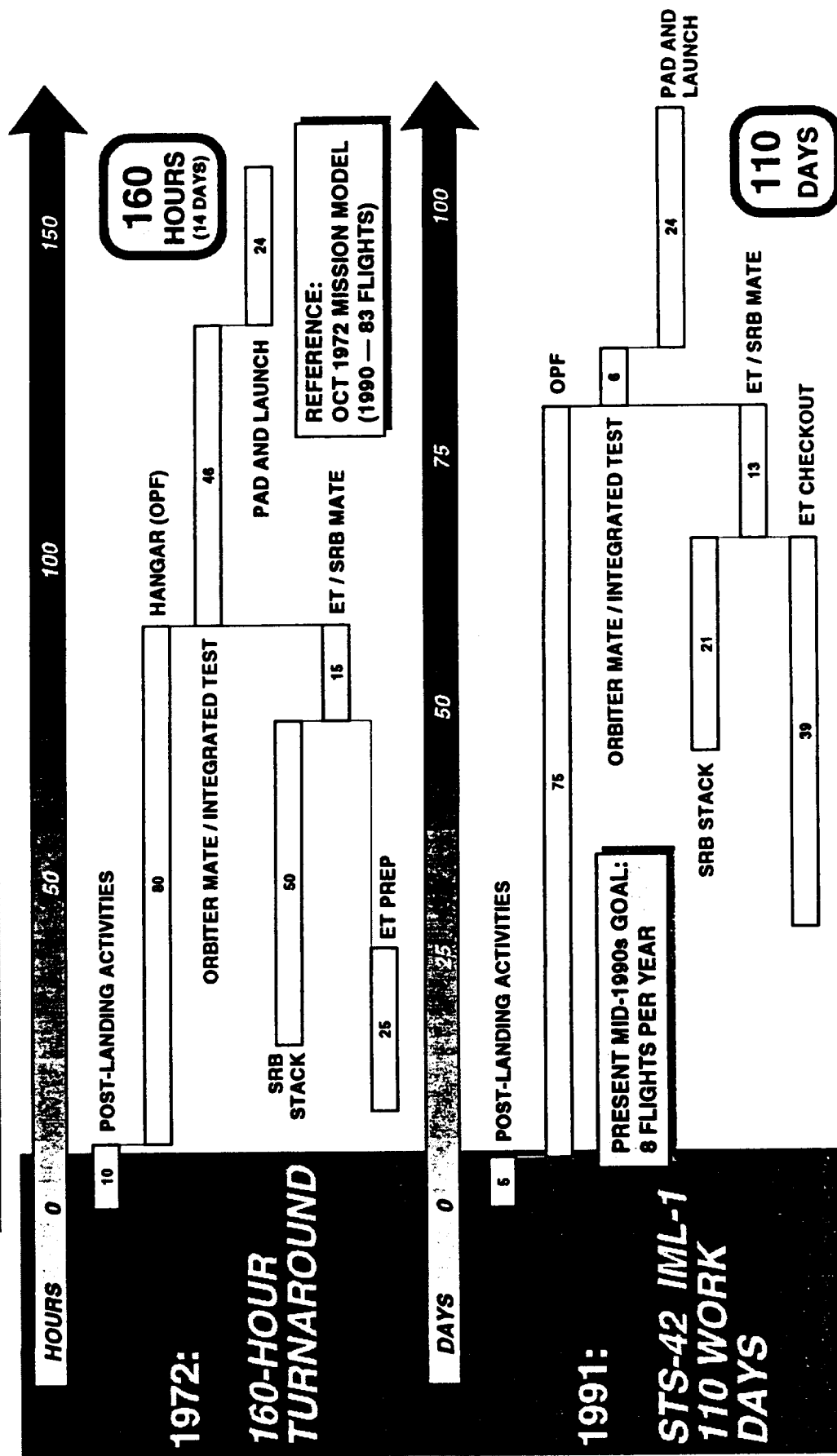
5 PAD PROCESSING



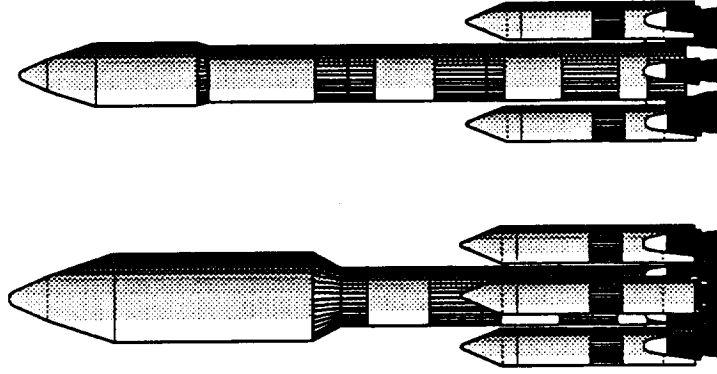
Space Exploration Initiative

LAUNCH SITE OPERATIONS AND FACILITIES

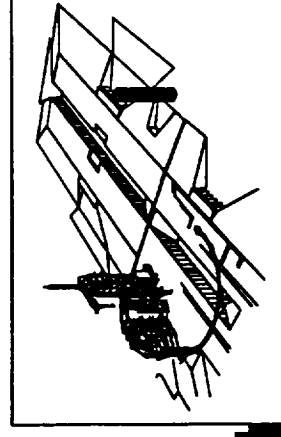
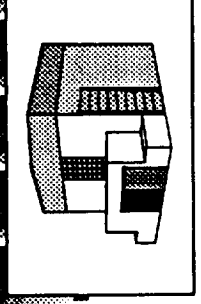
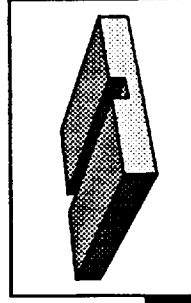
SHUTTLE TURNAROUND COMPARISON



LUNAR OUTPOST



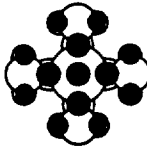
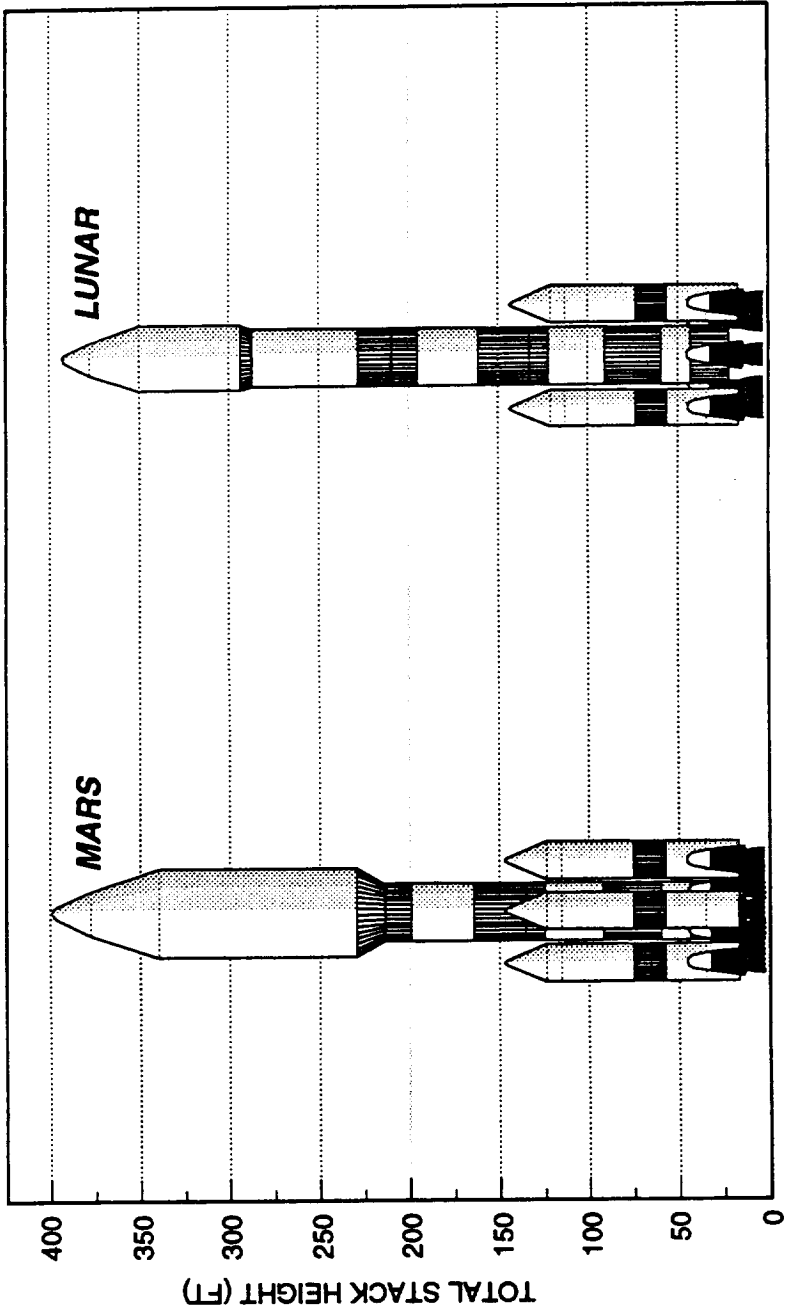
KSC LAUNCH SITE PROCESSING SATURN V-DERIVED SINGLE LAUNCH VEHICLES



7 MAY 1992

LUNAR OUTPOST

SATURN-DERIVED SINGLE LAUNCH VEHICLES



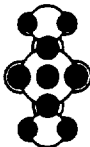
SATURN V S-IC 1st STAGE
SATURN V S-II 2nd STAGE
4X2 F-1A BOOSTERS
KICKSTAGE

GLOW

13.6 Mib

LEO PAYLOAD

230 mt



SATURN V S-IC 1st STAGE
SATURN V S-II 2nd STAGE
2X2 F-1A BOOSTERS
1 J-2S TLI STAGE

GLOW

10.2 Mib

POST TLI
PAYLOAD

93 mt

PRELIMINARY

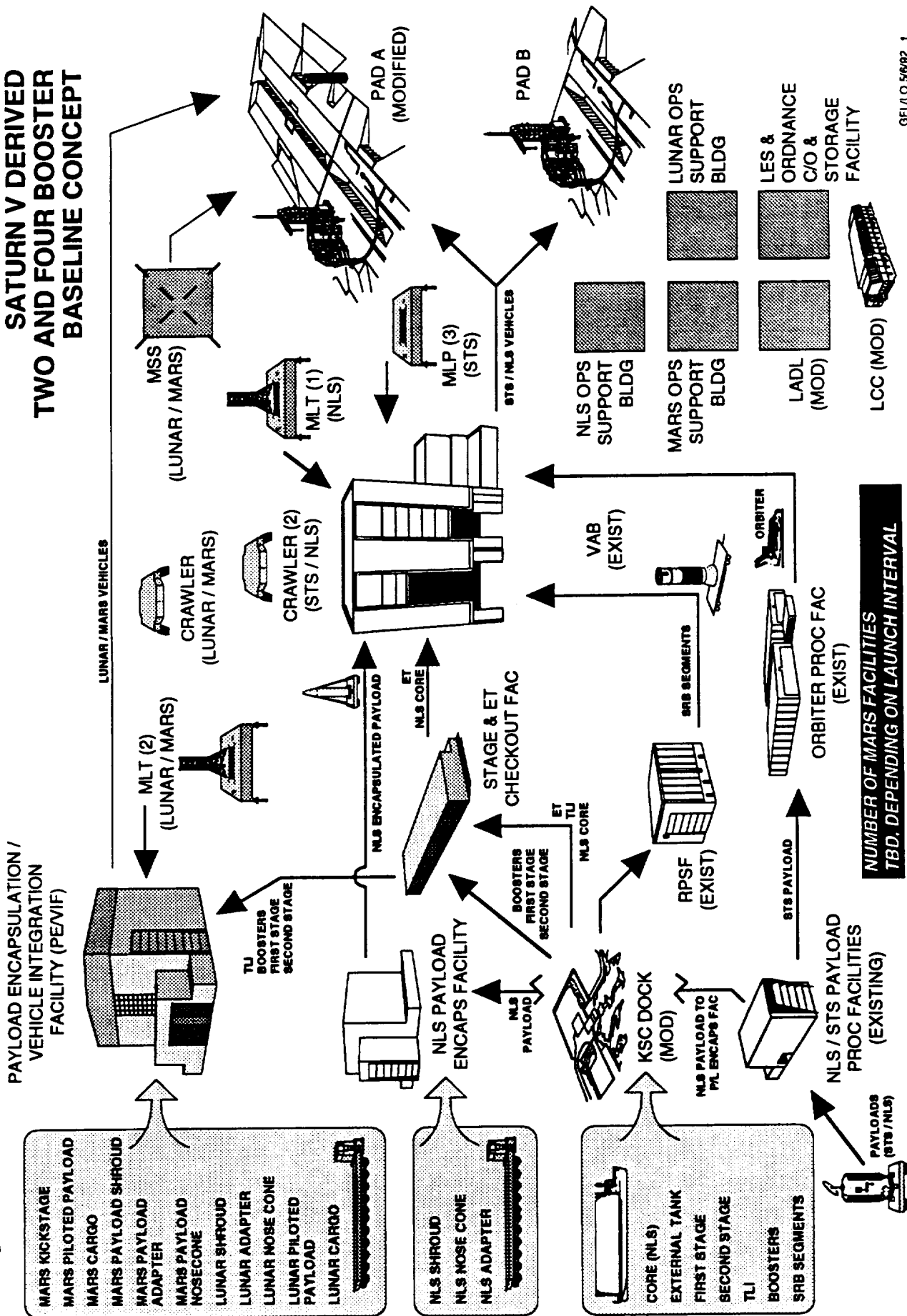
LAUNCH PROCESSING SCENARIO

LUNAR / MARS SINGLE LAUNCH

SATURN V DERIVED

TWO AND FOUR BOOSTER

BASELINE CONCEPT

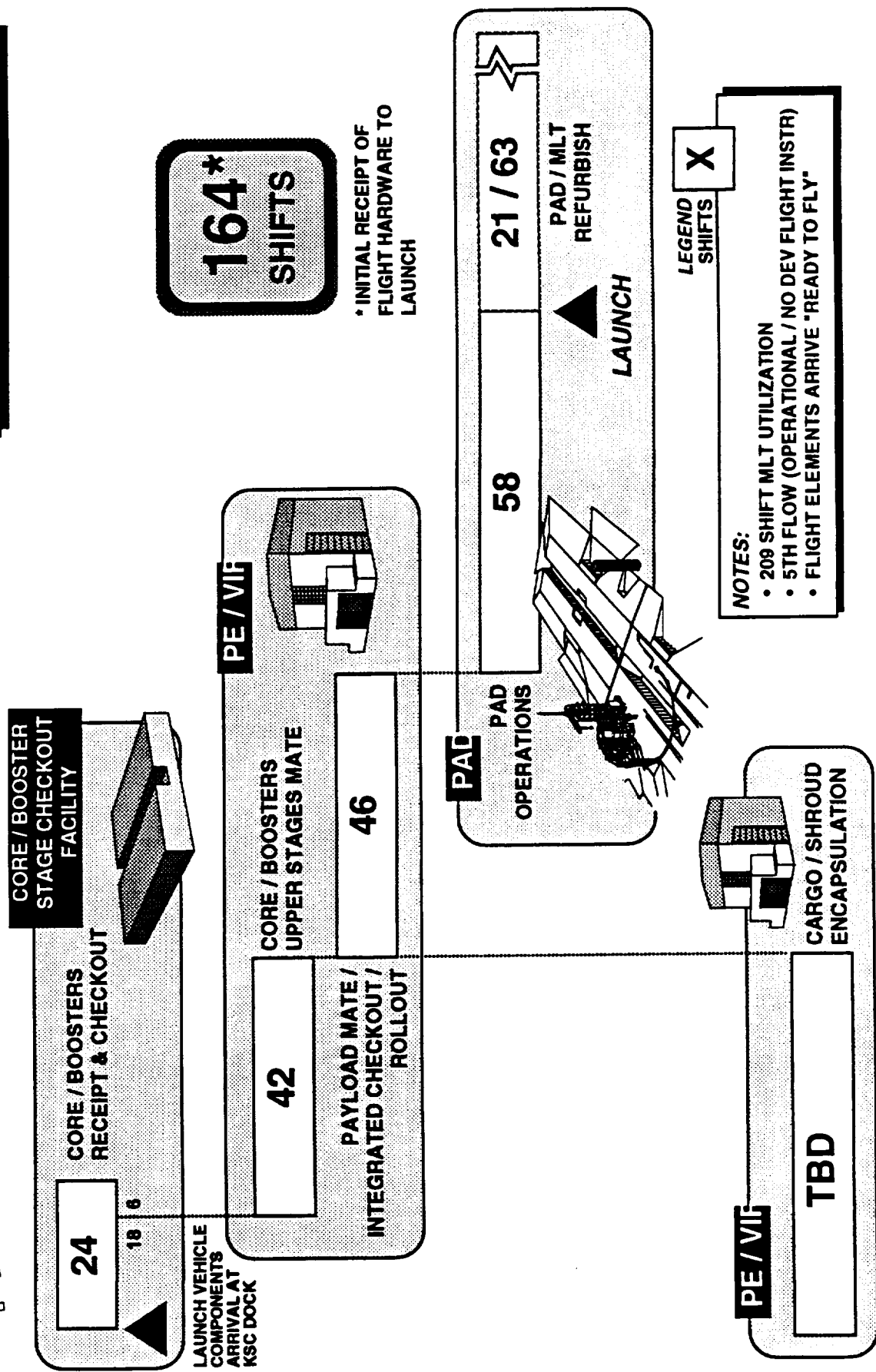


LUNAR OUTPOST

TOP LEVEL PROCESSING TIMELINE — OPERATIONAL

PRELIMINARY

SATURN V-DERIVED HLLV
2 LOX / RP-1 BOOSTERS

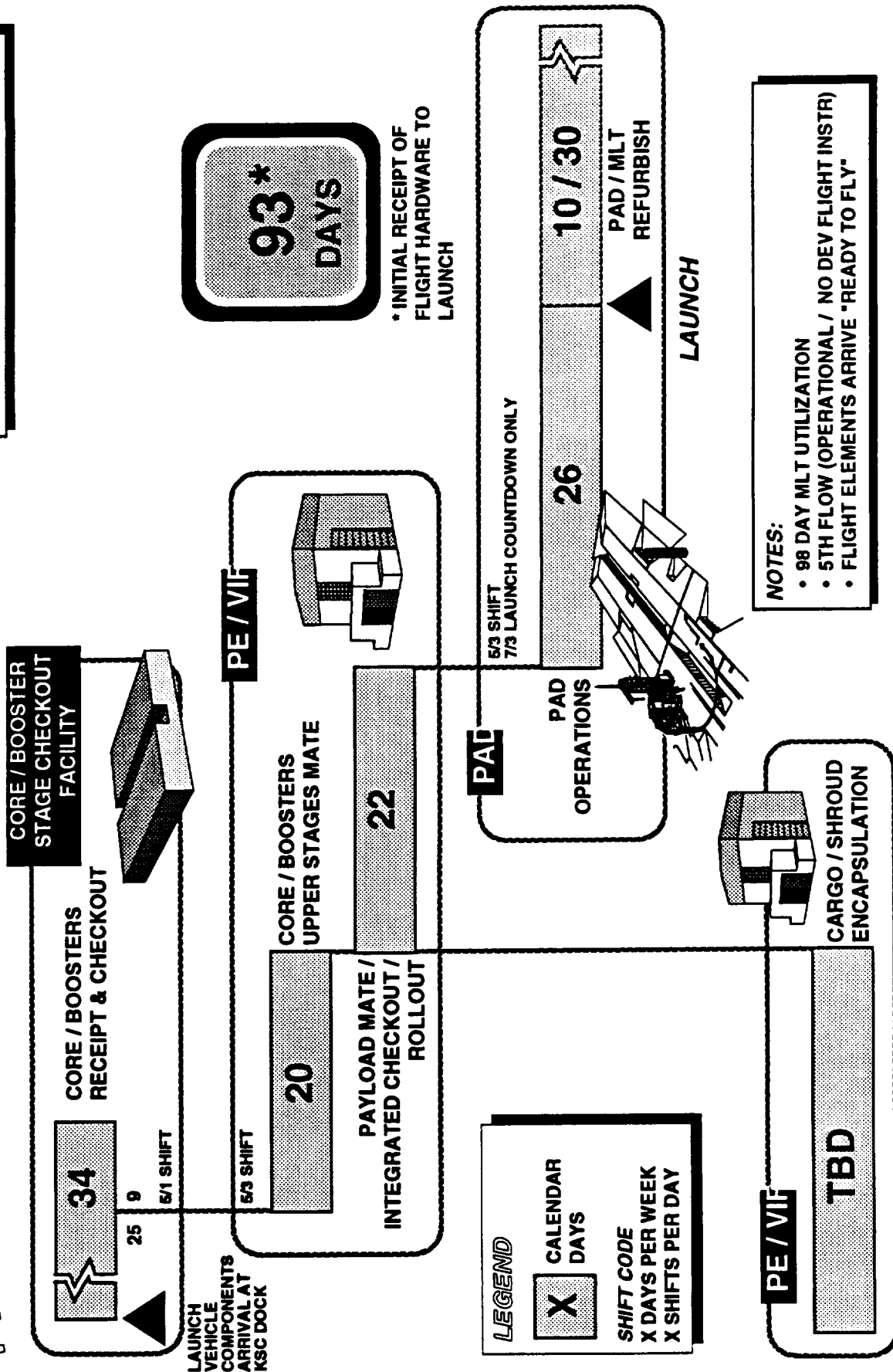


LUNAR OUTPOST

TOP LEVEL PROCESSING TIMELINE — OPERATIONAL

PRELIMINARY

SATURN V-DERIVED HLLV
2 LOX / RP-1 BOOSTERS



LUNAR OUTPOST

VEHICLE INTEGRATION TIMELINE

PRELIMINARY

SATURN V-DERIVED HLLV
2 LOX / RP-1 BOOSTERS

MLP READY TO
STACK

11 CORE TRANSPORT
& STACK

11 BOOSTER 1
TRANSPORT & MATE

11 BOOSTER 2
TRANSPORT & MATE

7 2ND STAGE
TRANSPORT & MATE

7 TLI STAGE TRANSPORT
& MATE

21 INTEGRATED LAUNCH
VEHICLE TEST & CLOSEOUT

7 ENCAPSULATED PAYLOAD (SPACE
CRAFT) TRANSPORT & MATE

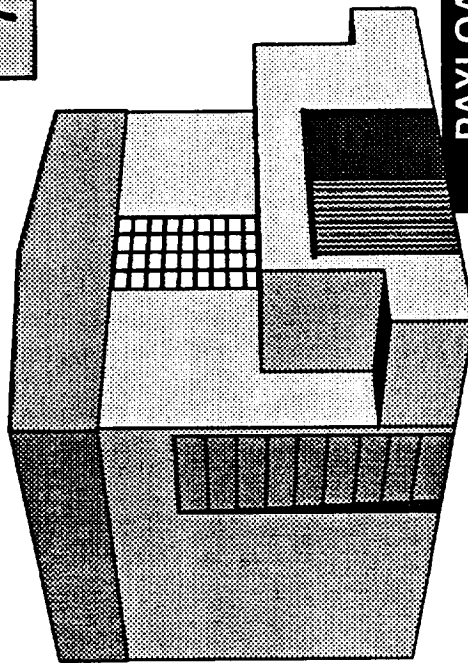
12 INTEGRATED LAUNCH VEHICLE /
SPACE CRAFT TEST & CLOSEOUT

3 LES TRANSPORT &
MATE

3 ROLLOUT
PREPS

LAUNCH VEHICLE
READY FOR ROLLOUT

PAYLOAD ENCAPSULATION /
VEHICLE INTEGRATION FACILITY
(PE / VIF)



X X SHIFTS
TYPICAL

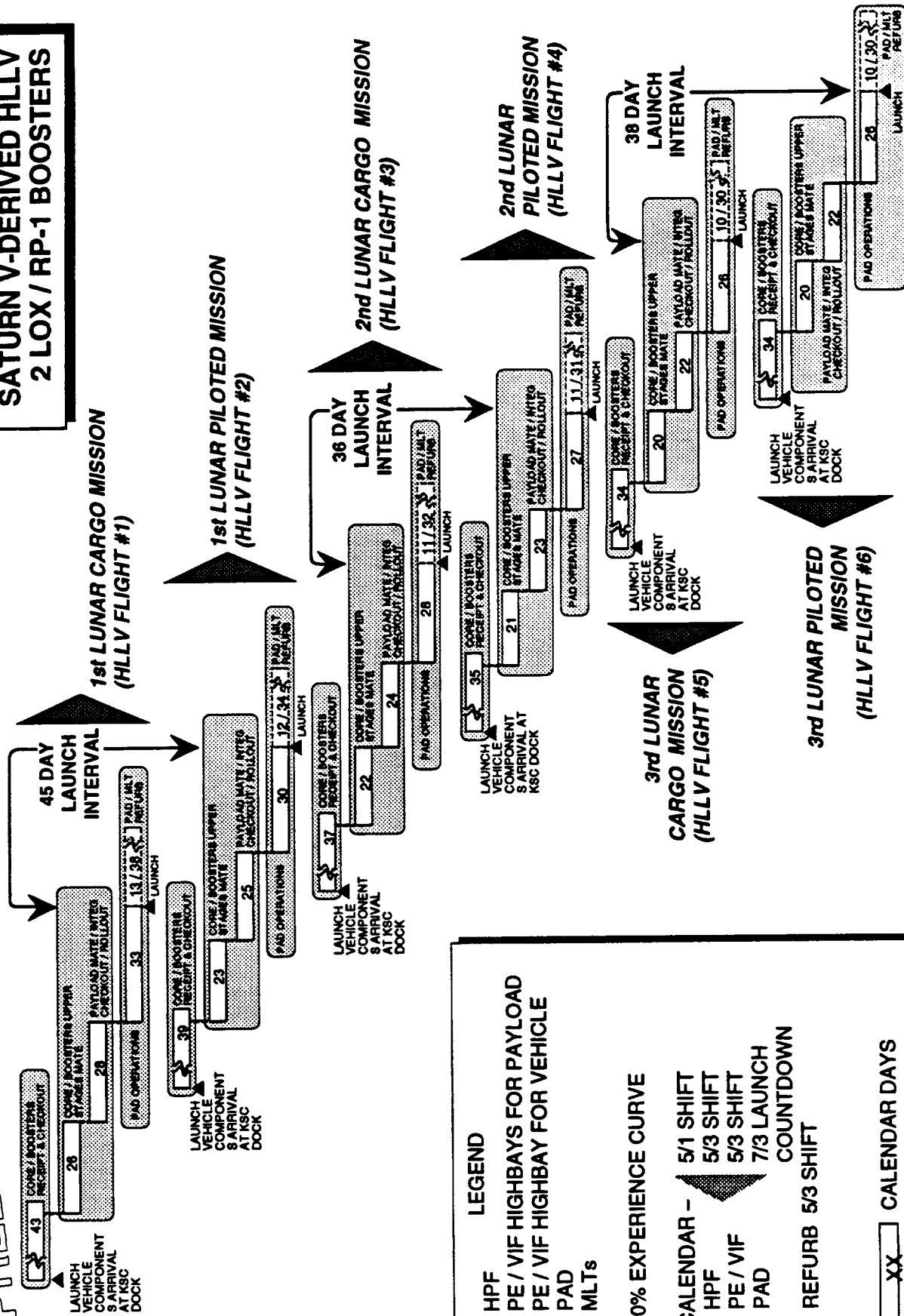
CONCEPTUAL
PROCESSING FLOW
SUMMARY LEVEL
PILOTED VEHICLE ONLY

LUNAR OUTPOST

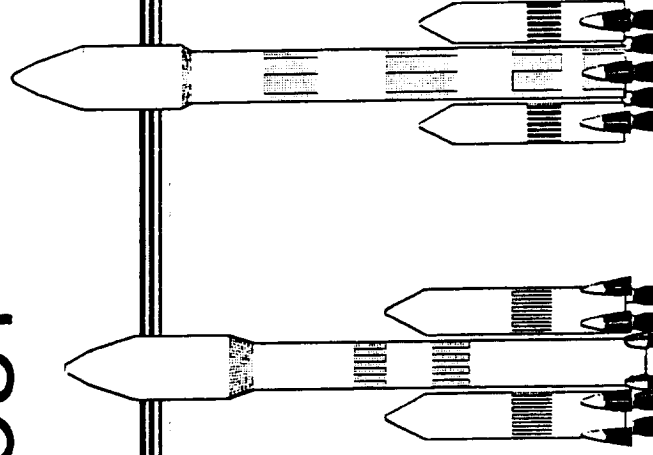
PRELIMINARY

MULTI-LAUNCH TIMELINE

SATURN V-DERIVED HLLV
2 LOX / RP-1 BOOSTERS



FIRST LUNAR OUTPOST STUDIES



DRAFT
4/24/92

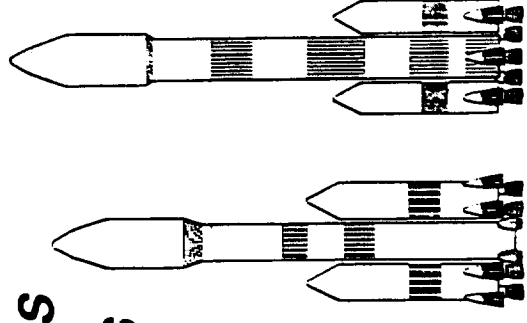
INTERIM REPORT
LSOC ADVANCED PROGRAMS
APRIL 1992

 **Lockheed**
Space Operations Company

- PROPOSED ETO LAUNCH VEHICLES
- GROUND PROCESSING SCENARIOS
- LAUNCH PROCESSING TIMELINES
 - SINGLE LAUNCH MISSION CONCEPT
- SATURN V-DERIVED CORE*
- NLS-DERIVED CORE*
- DUAL LAUNCH MISSION CONCEPT
- NLS-DERIVED CORE LOX / RP-1 BOOSTERS*
- NLS-DERIVED CORE LOX / LH₂ BOOSTERS*

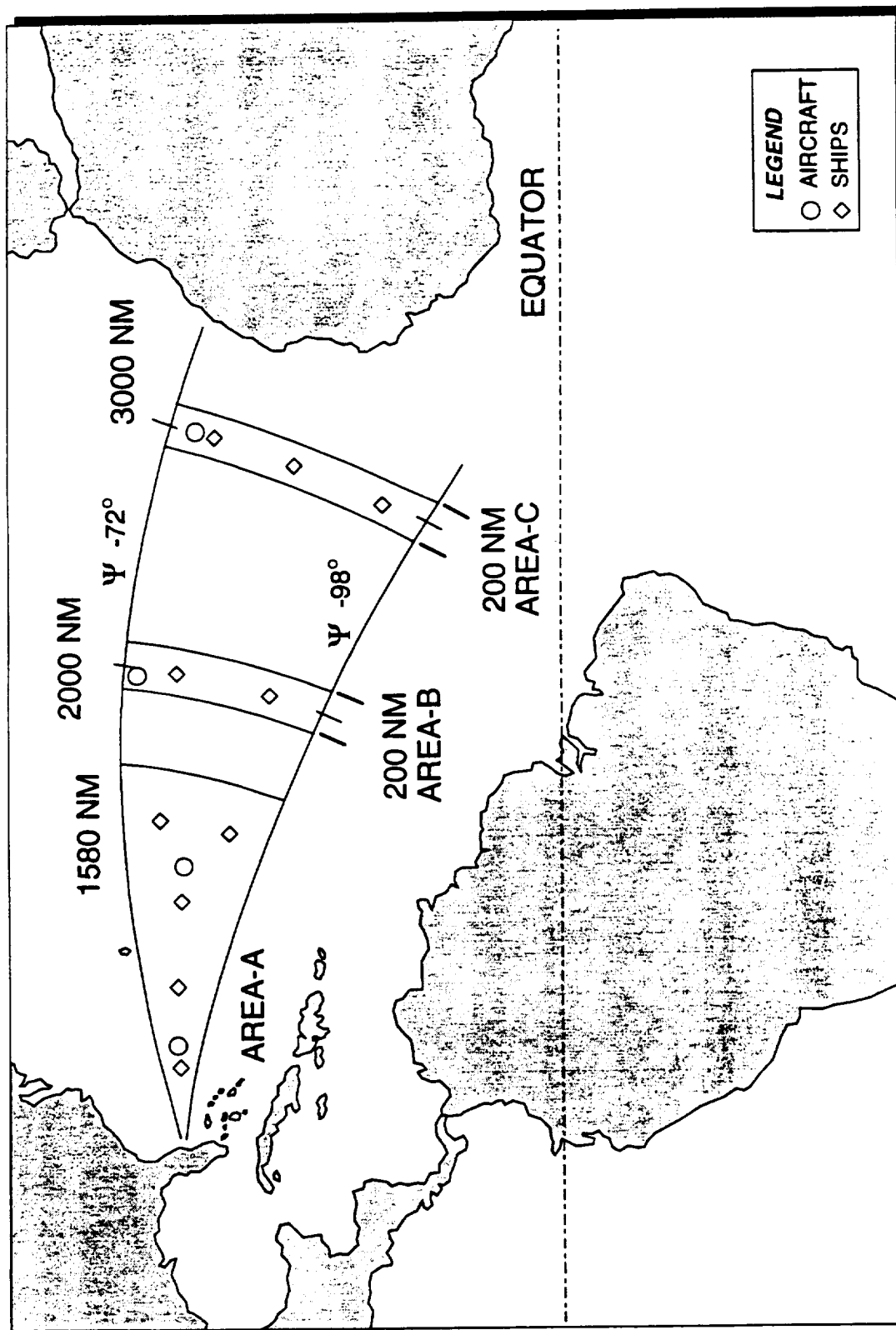
■ EXPERIENCE CURVES

■ RECOVERY ELEMENT ASSESSMENTS



LUNAR OUTPOST

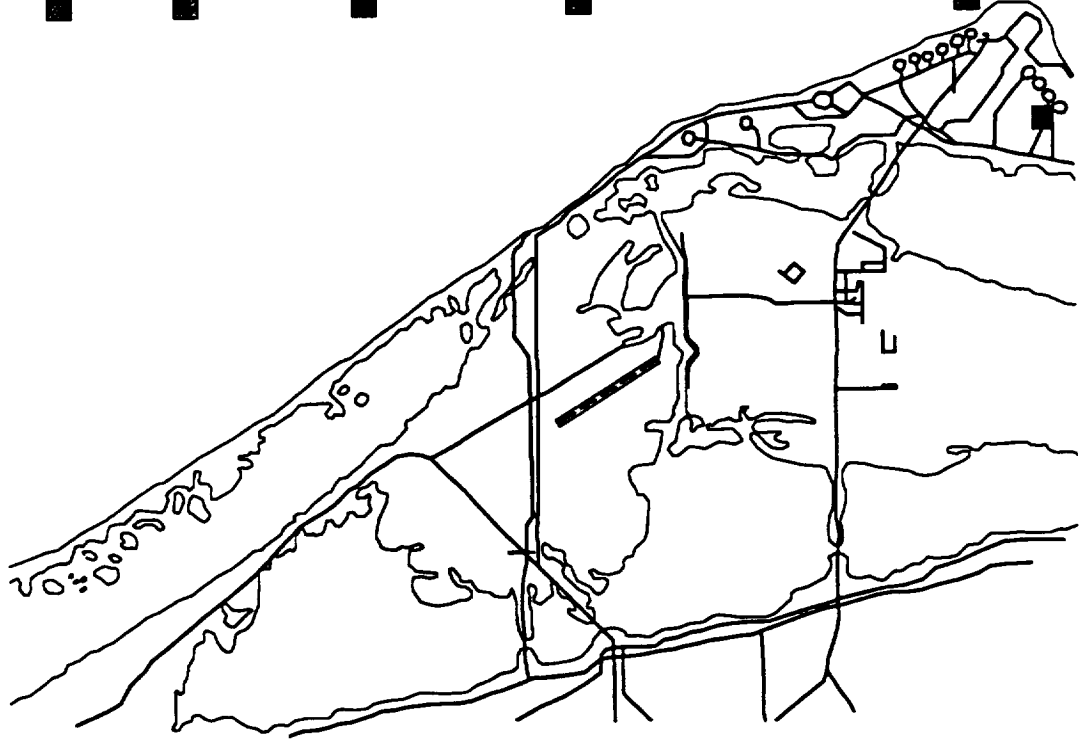
APOLLO VARIABLE LUNAR LAUNCH AZIMUTH LAUNCH ABORT AREAS



REF: PROJECT APOLLO PRELIMINARY RECOVERY REQUIREMENTS

LUNAR OUTPOST

OCEAN RECOVERY



- ASSUME WATER LANDER @ 50 TO 200 NMI OFF FLORIDA COAST
- JETTISON PROPULSION MODULE PRIOR TO SPLASHDOWN TO MINIMIZE TOXIC HAZARDS
- SEARCH AND RESCUE (SAR) REQUIREMENTS DERIVED FROM DDMS MODE VIII (ASTRONAUT BAILOUT SAR) EXERCISE FOR STS (JULY 1990)
- RESOURCES REQUIRED:
 - 3 HH-3 HELICOPTERS
 - 2 HC-130 TANKER AIRCRAFT (PLUS 1 STANDBY HC-130)
 - 6 ZODIAC RAFTS (3 ON EACH HC-130)
 - 4 MA-2 SURVIVAL KITS (2 ON EACH HC-130)
 - 27 PARARESCUE SPECIALISTS (3 PER HH-3; 9 PER HC-130)
 - 1 E2-C AIRCRAFT (PLUS ONE STANDBY E2-C)
 - 1 COAST GUARD CUTTER OR 1 US NAVY SHIP
 - MEDICAL, SURVEILLANCE, PUBLIC AFFAIRS SUPPORT
- JULY 1990 MODE VIII RESCUE TOOK 3 HOURS

LUNAR OUTPOST

LAND LANDER CONCEPT

PERSONNEL

- SAFETY REPRESENTATIVES
- VEHICLE TECHNICIANS
- FIRE SERVICES
- EMERGENCY MEDICAL SERVICES
- SECURITY
- LIFE SUPPORT PERSONNEL
- WEATHER
- ENVIRONMENTAL HEALTH

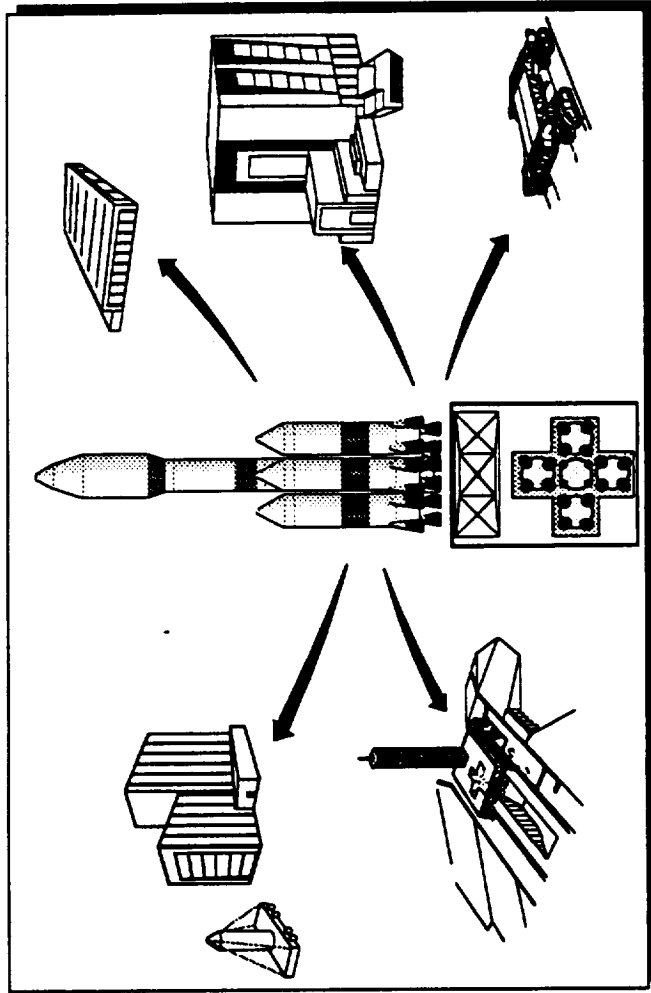
EQUIPMENT

- SCAPE (SAFETY REPS AND TECHS)
- SCOTT / RANGER AIR PACKS W/ COMM
- TOXIC VAPOR DETECTORS (FUEL AND OXIDIZER)
- FLAMMABILITY CART FOR MMH, HYDRAZINE, AND AMMONIA
- MOBILE WIND MACHINE
- SAFETY NETWORK PORTABLE RADIOS
- BREATHING ESCAPE UNITS
- FREON-21 DETECTOR
- DRAEGERS WITH N2O4, HYDRAZINE, AND NH3 TUBES
- TOW VEHICLES
- WHITE ROOM
- FIRE TRUCKS

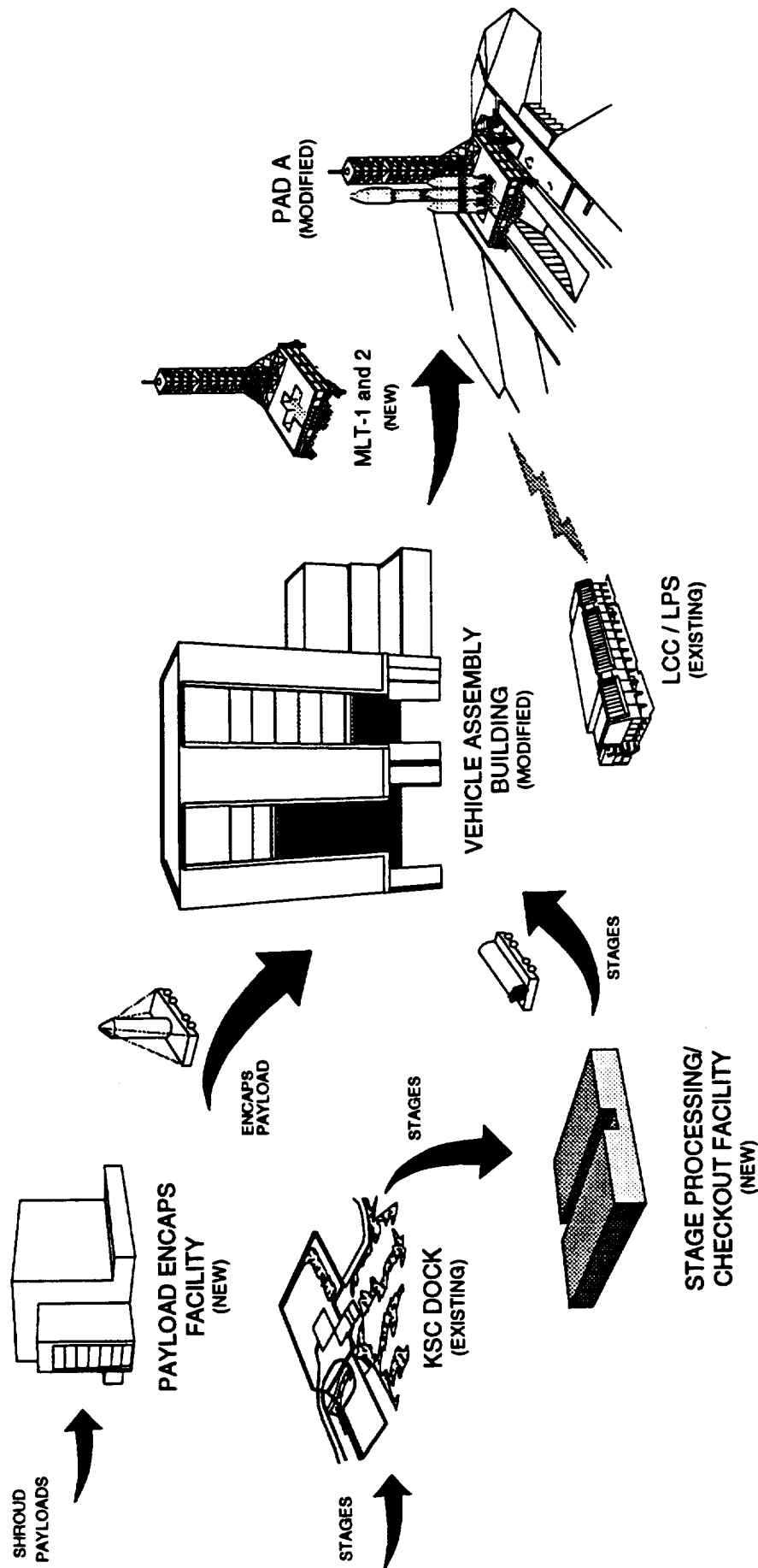
SINGLE LAUNCH LUNAR/MARS VEHICLE

LAUNCH SITE FACILITY PRELIMINARY ASSESSMENT

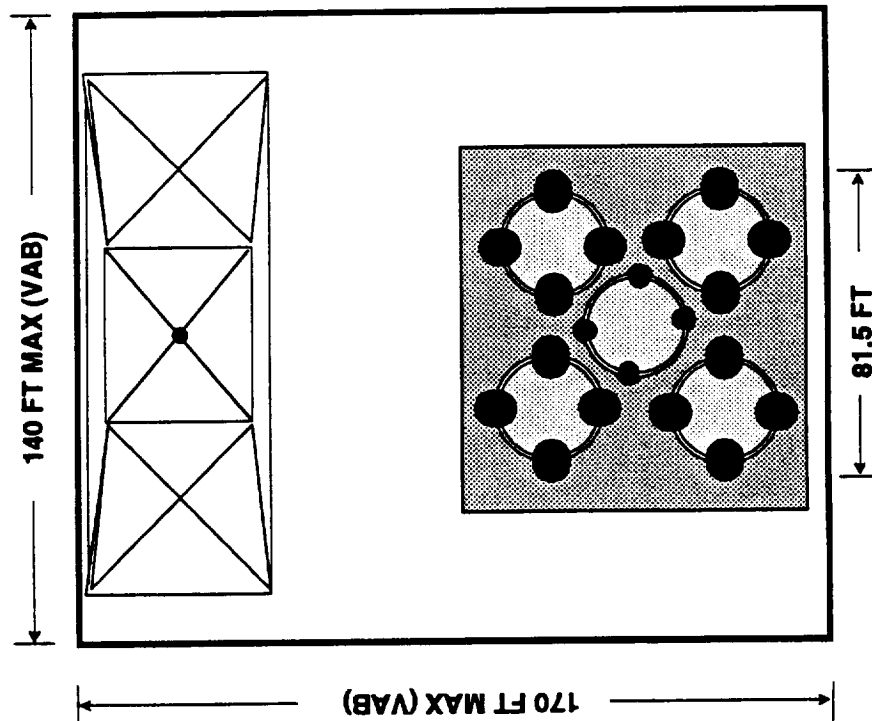
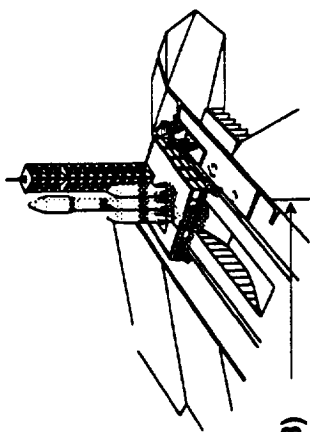
MARCH 1992



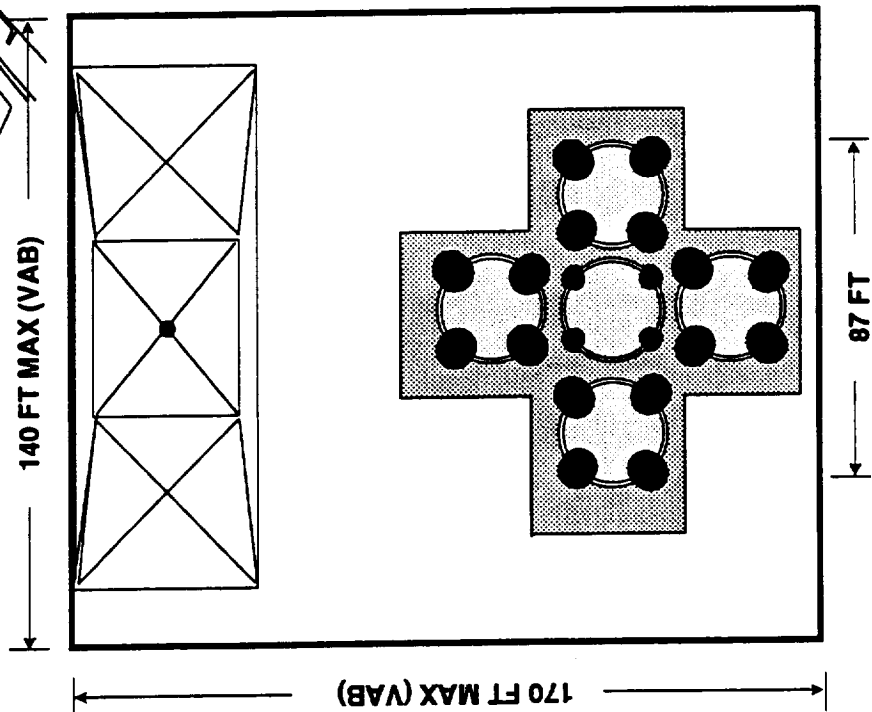
SINGLE LAUNCH VEHICLE LAUNCH PROCESSING SCENARIO



SINGLE LAUNCH VEHICLE VEHICLE ORIENTATIONS ON MLT

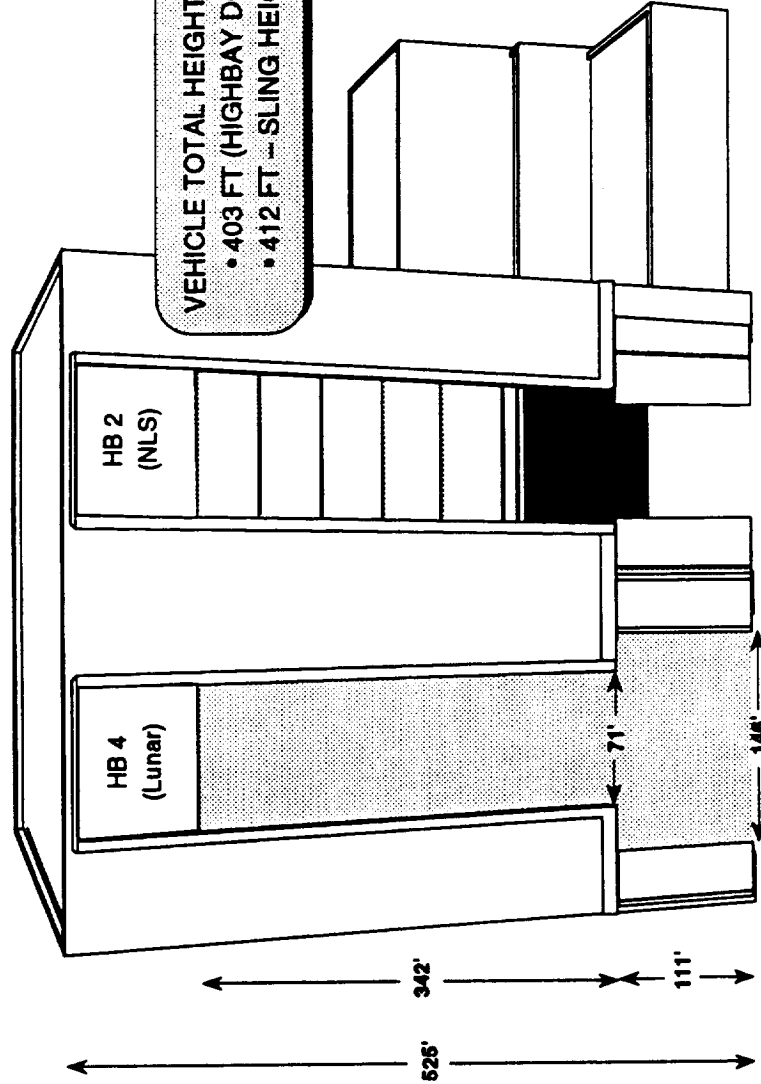
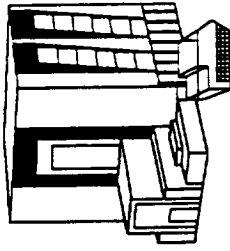


X ORIENTATION



PLUS ORIENTATION

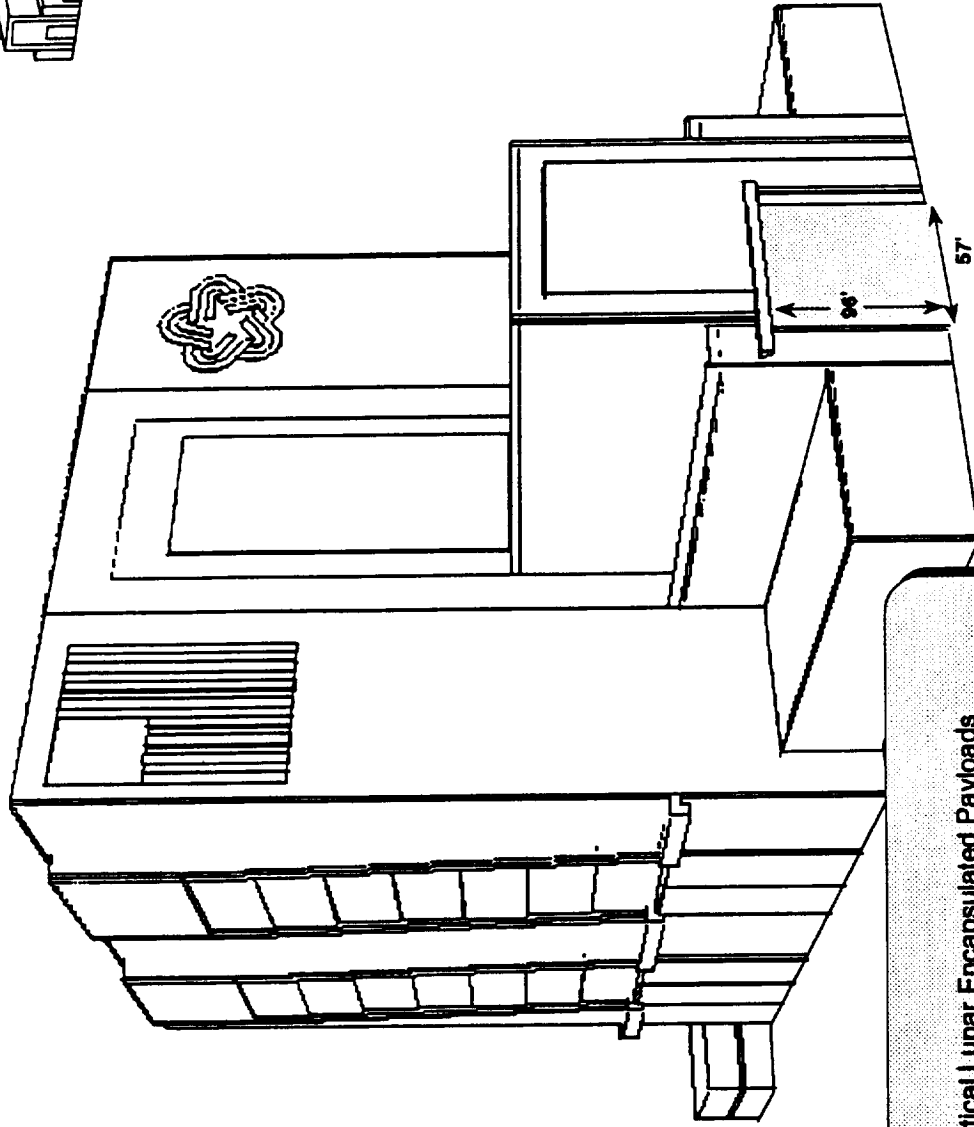
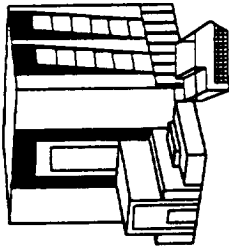
VAB HIGHBAY FACILITY LIMITATIONS HIGHBAYS 2 AND 4



NOTES:

- 1) VAB HIGHBAY HOOK HT. 462 FT (TOTAL DOOR OPENING HT. 453 FT)
- 2) VAB HIGHBAY/TRANSFER ISLE DIAPHRAM BEAM HT. 190 FT (BEAM TO HOOK CLEARANCE 272 FT)
- 3) CRAWLER & MLT HT APPROX. 50 FT

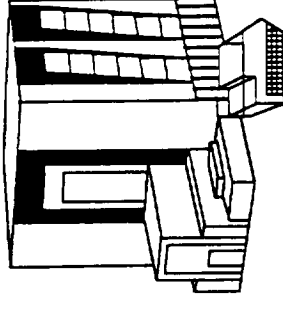
VAB Highbay Facility Limitations (cont) VAB South Transfer Isle



NOTES:

- 1) Insufficient clearance for 100 ft vertical Lunar Encapsulated Payloads
- 2) Insufficient clearance for 175 ft vertical Mars Encapsulated Payloads

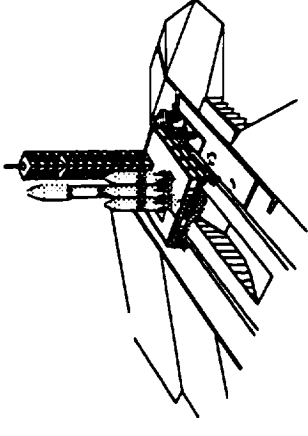
SINGLE LAUNCH VEHICLE LAUNCH SITE ISSUES (cont)



VAB

- VAB WILL BEST ACCOMMODATE HIGH ENERGY DENSITY (LOX/RP-1) BOOSTERS/CORE
- VAB LIMITS HEIGHT OF VEHICLE AND MOBILE LAUNCHER WITH TOWER
- SHARING VAB WITH SHUTTLE PRESENTS RISK — SRB STACKING
- ETs CHECKOUT CELLS (2) MUST BE MOVED FROM HIGHBAY 2 OR 4
- TRANSFER ISLE NORTH OR SOUTH DOOR MUST BE MODIFIED FOR ENCAPS PAYLOADS
- HIGHBAY UPPER DOORS MUST BE WIDENED FOR 4 BOOSTER VEHICLES
(IS THIS STRUCTURALLY FEASIBLE?)
- A 4 BOOSTER VEHICLE/MOBILE LAUNCHER WITH TOWER MAY NOT BE FEASIBLE

SINGLE LAUNCH VEHICLE LAUNCH SITE ISSUES (cont)



PAD

- PAD PRESENTS POTENTIAL MIXED FLEET BOTTLENECK AND SINGLE POINT FAILURE

- 1) LUNAR, MARS, NLS, AND STS SHARE LC-39A?
- 2) FLIGHT INTERVAL BETWEEN 3 PROGRAMS
- 3) PAD DOWN-TIME FOR 2-3 NEW PROGRAM MODIFICATIONS

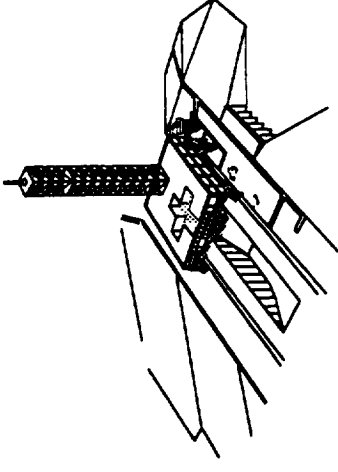
- SIGNIFICANT MODIFICATIONS REQUIRED TO SUPPORT LUNAR/MARS

- 1) PROPELLANT TANK FARMS
- 2) FLAME DEFLECTORS AND TRENCHES
- 3) WATER DELUGE SYSTEM

- MAJOR TRADE STUDIES NEED TO BE MADE FOR ANY NEW PROGRAM USING LC-39

- 1) LAUNCH ENVIRONMENT EFFECTS ON EXISTING STRUCTURES (RSS, FSS, etc)
- 2) FIXED SERVICE STRUCTURE (FSS) AT PAD VERSUS MOBILE LUT
- 3) FLAME DEFLECTOR/FLAME TRENCH SIZING

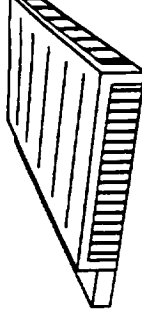
SINGLE LAUNCH VEHICLE LAUNCH SITE ISSUES (cont)



MOBILE LAUNCHER W/ TOWER

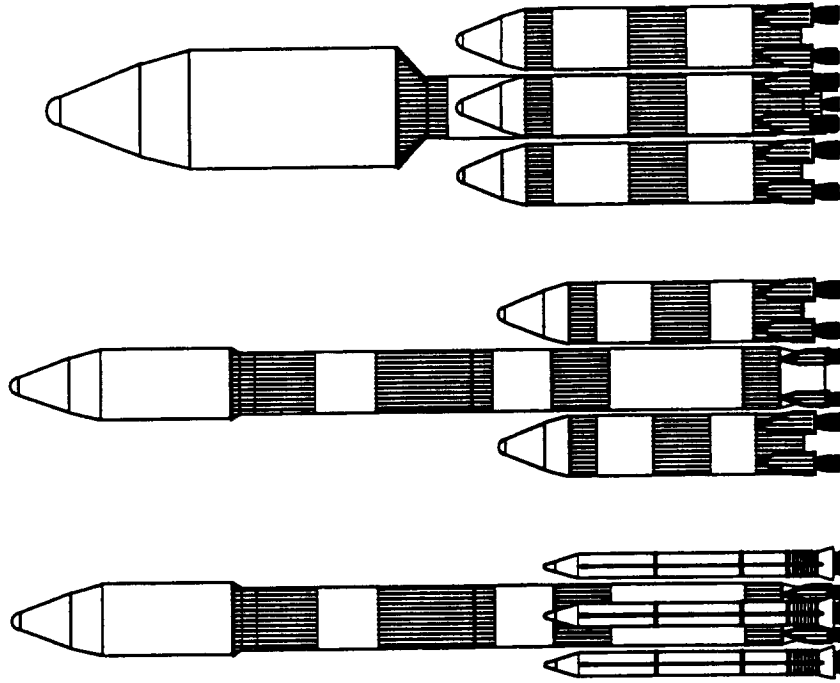
- TRADE NEEDS TO BE MADE OF TOWER AT THE PAD VS. ON THE LAUNCHER
- IS 4.5-6 DEG. VEHICLE-TO-TOWER DRIFT CLEARANCE VALID?
- MOBILE LUT WILL REQUIRE EXTENDABLE LIGHTNING PROTECTION (45 DEGREE CONE) AT 550-600 FT ELEVATIONS
- HIGH ENERGY DENSITY BOOSTERS/CORE RESULT IN:
 - 1) REDUCED TOWER AND MOBILE LAUNCHER WEIGHT, HEIGHT, AND AREA
 - 2) REDUCED ON-PAD WIND LOADING OF VEHICLE W/ ATTENDANT SNUBBERS REQ
- THE MOBILE TOWER MAY CONSUME MORE MLT DECK AREA THAN THE VEHICLE FOR TALL NARROW CONFIGURATIONS
- MLT IS THE CRITICAL PATH FACILITY

SINGLE LAUNCH VEHICLE LAUNCH SITE ISSUES (cont)



STAGE PROCESSING FACILITY

- ASSUME VEHICLE ELEMENTS ARRIVE READY-TO-FLY, THEREFORE SHORT C/O TIMES
- FACILITY WILL MAINLY BE USED FOR SURGE, STORAGE, AND CONTINGENCIES
- IF VAB IS USED FOR INTEGRATION, MOVE SHUTTLE ETs TO THIS FACILITY
- SHOULD BE SIZED TO ACCOMMODATE LUNAR AND MARS PROGRAMS
- IF NLS-DERIVED CORE IS USED, WILL IT BE CHECKED HERE OR AT NLS CSPF AT CANAVERAL?



5. Concluding Remarks

